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AFRPL TR-72-45

FINAL REPORT

ORBIT-TO-ORBIT SHUTTLE ENGINE DESIGN STUDY

Contract F04611-71-C-0040

BOOK 1

W. P. Luscher, et. al.
Aerojet Liquid Rocket Company
Sacramento, California

May 1972

Air Force Rocket Propulsion Laboratory
Edwards Air Force Base, California

Best Available Copy

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Headquarters, Air Force Flight Test Center
Rocket Propulsion Laboratory
Edwards Air Force Base, California

[redacted]

FOREWORD

This report presents the work accomplished on Contract F04611-71-C004, the Orbit-to-Orbit Shuttle Engine Design Study (OOS) over the period from 1 March 71 to 1 December 1971. The program was admistered by the Procurement Division of the Directorate of Material, Edwards Air Force Base, Edwards, California. The technical project manager at the Rocket Propulsion Laboratory, Edwards, California was Mr. L. Tepe. Mr. Werner P. Luscher directed the study effort for Aerojet Liquid Rocket Company.

This report is contained in 4 books described as follows:

Book 1: Parametric Cycle Study

Book 2: 25K lb Engine Design

Book 3: 25K lb Engine Maintenance, Development Plans, Cost Estimates and 10K lb Engine Design

Book 4: Appendices

This technical report has been reviewed and is approved.

L. E. Tepe
Project Manager

ABSTRACT

This report presents the analytical design of propulsion systems utilizing LOX/Hydrogen propellants to be used as the propulsion for the Orbit to Orbit Space Vehicle of 65,000 lb lift-off weight.

The report contains the evaluation of various engine cycles in the thrust range of 8,000 lb to 50,000 lb thrust for performance, weight and envelope culminating in the cycle selection and detail design of a 25,000 lb and 10,000 lb thrust engine. The engine concepts are described in sufficient detail to obtain reliable engine weight, performance, envelope information and methods of engine control. The impact of various engine design requirements were evaluated. The engines are designed to be reusable and capable of starting in the idle mode operation.

The technology requirements for meeting the engine design and operating requirements are identified.

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NOMENCLATURE

UNITS:

°F	Degrees Farenheit
ft	Feet
GPM	Gallons per Minute
hr	Hours
HP	Horsepower
in.	Inches
lb	Pounds
mm	Millimeters
psi	Pounds per Square Inch
rpm	Revolutions per Minute
°R	Degrees Rankine
sec	Seconds

SYMBOLS:

A/R	Area Ratio
CR	Contraction Ratio
DN	Bearing (Bore Diameter mm x rpm)
ϵ	Area Ratio
F	Thrust
I_s	Specific Impulse
L'	Chamber Length
MR	Mixture Ratio
M	Mach Number
MRD	Mixture Ratio Distribution
Ns	Specific Speed
N_f	Number of Cycles
P_c	Chamber Pressure
PVC	Pressure Volume Compensated
SF	Safety Factor
S	Suction Specific Speed
T_{T_i}	Turbine Inlet Temp
T_{FD}	Fuel Pump Discharge Temp
T	Chamber Hot Wall Temp
ΔT_W	Chamber Wall Temp Gradient
T_B	Bulk Temperature
V	Velocity
$\dot{W}$	Weight Flow Rate
W_{BO}	Burnout Weight
η	Efficiency

GLOSSARY:

ALRC	Aerojet Liquid Rocket Company
AGCarb	Carbon Cloth Material
CJKT	Preburner & Turbine By-pass
ERE	Energy Release Efficiency
FTPA	Fuel Turbopump Assembly
FDV	Fuel Discharge Valve
FRHG	Fuel Rich Hot Gas
FPBYV	Fuel Preburner & Turbine By-Pass Valve
FFC	Final Flight Configuration
Hz	Frequency
LRU	Line Replaceable Unit
N_{TF}	Fuel Turbine rpm
N_{TO}	Oxidizer Turbine rpm
N_{TFE}	Fuel Low Speed Inducer rpm
N_{TOE}	Ox Low Speed Inducer rpm
NPSH	Net Position Suction Head
NPSP	Net Positive Suction Pressure
OTPA	Oxidizer Turbopump Assembly
OPBV	Oxidizer Preburner Valve
OSCV	Oxidizer Mixture Ratio Control Valve
PL	Payload
PFC	Preliminary Flight Configuration
TCA	Thrust Chamber Assembly
TGJD	Temp at FRHG Injector Inlet
UTMO	Max Oxidizer Turbine Tip Speed
UTMF	Max Fuel Turbine Tip Speed

SECTION I

INTRODUCTION

Contract F04611-71-C-0040, the Orbit-to-Orbit Shuttle Design Study (OOS), was awarded to the Aerojet Liquid Rocket Company (ALRC) on 1 March 1971. The program was administered by the Procurement Division of the Directorate of Material of the Edwards Air Force Base, Edwards, California. The technical direction originated at the Rocket Propulsion Laboratory, Edwards, California.

The basic objective of the OOS Program was to provide analytical design performance and cost data for advanced O_2/H_2 engine systems that could be used in a variety of high energy, space maneuvering missions.

The near term application of the study was the main propulsion system of the Orbit-to-Orbit Shuttle or Space Tug. The Earth-to-Orbit (EOS) portion of the Space Transportation System will be able to place large payloads into low-earth orbits in the late 1970's. The vehicle payload carried by the EOS is the Orbit-to-Orbit Shuttle. The OOS must be capable of transferring payloads from the low-earth orbit of the EOS to other orbits. To perform these types of missions efficiently, the OOS vehicle system must be designed for high performance, multiple re-starts, long-term space storability, long life, reusability, low maintenance, and low cost.

The study program consisted of five discrete tasks which are described below:

1. Consideration of the impact of varying the maximum thrust level between 8,000 and 50,000 lb thrust, with the objective of selecting the optimum engine design and turbine drive cycle for each thrust regime. The turbine drive cycles to be considered included the gas generator, staged combustion, staged combustion bleed, expander, coolant gas top-off and combustion gas top-off cycles.

2. The analysis and preliminary design of a 25,000 lb thrust engine.

3. The effects of varying the engine mixture ratio and turbopump NPSH for 8,000, 15,000 and 25,000 lb thrust engine systems were determined.

4. Study of the effects of design variations on the 25,000 lb engine.

5. The development of plans and costs for a Demonstrator Engine Program, a development program, and production of three versions of the Task I engine. The three versions were ground based reusable, space based reusable and expandable.

The technical effort associated with the first five tasks was completed on 1 September 1971, coincident with the award of a contract extension for Phase VI studies.

I, Introduction (cont.)

The studies were directed to the design of a 10,000 lb thrust engine utilizing the results of the previous five tasks to the maximum extent practicable.

SECTION II

SUMMARY

The scope of the OOS Engine Design Study included the selection of a basic engine cycle resulting in the largest payload capability from 8K to 50K thrust for the Orbit-to-Orbit Mission within the given envelope and life constraints. Based on the results of the study, the basic engine cycle and design point was selected which formed the basis for a 25K and 10K thrust engine design. The turbine drive cycles studied were: staged combustion, staged combustion bleed, expander, gas generator, coolant tap-off and hot gas tap-off. In arriving at the design concepts, only component technology which would be proven during an advanced development program was considered.

The 25K thrust engine design effort consisted of the definition of the engine concept and configuration for a reusable, throttleable space engine including the complete characterization of the operational characteristics, e.g., off-design, chilldown, start transient, and interface definition. The interface definition included envelope, suction line size, gimbal actuator, purge requirement, autogenous pressurization system, and preconditioning requirements.

The basic engine design was evaluated with respect to the impact of changing design requirements, on engine configuration, performance, weight, and envelope.

For the purpose of establishing engine cost figures, a tentative engine development plan was established including a demonstrator engine phase and an engine development phase which preceed the engine production program. The cost study also treated the impact of engine thrust level, cycle, and design requirements.

The technical approach for the engine cycle selection was to establish the optimum chamber pressure for each engine cycle and for each thrust level. This chamber pressure had to satisfy simultaneously the 300 cycle engine life requirement, envelope restriction, and engine power balance requirement.

The baseline 10K thrust engine design is essentially a scaled version of the 25K engine with minor differences in the turbopump and autogenous systems for pump assisted idle mode operation. In addition an alternate in line design utilizing a LOX pump gear drive was established.

As the basic engine nozzle configuration, a minimum length Rao nozzle contour was assumed, and the corresponding thrust chamber and engine performance was determined utilizing the JANNAF thrust chamber performance prediction method. An injector performance efficiency of ERE = 99% of theoretical was assumed.

To obtain engine weights, a baseline engine design point was selected and appropriate layouts made for each cycle. The engine weights obtained from these layouts were selected as functions of engine thrust, chamber pressure mixture ratio, area ratio and NPSH.

Conclusion and Recommendations

The practical maximum chamber pressure, meeting the 300 cycle life, was established utilizing existing material fatigue life test data for Zirconium Copper material. However, available data is very sketchy, forcing large safety factors in the thrust chamber design.

With the selected thrust chamber pressure and available stowed engine length, the payload capability for each cycle was determined in the thrust range from 8K to 50K lb thrust for both retractable and fixed nozzle configuration. The result of this analysis indicated that the staged combustion cycle results in the highest payload capability within the thrust range considered. Consequently, was selected as the engine cycle for the OOS 25K and 10K thrust engine designs.

The basic engine design for both engines features fixed minimum Rao nozzle configuration with hydrogen regeneratively cooled thrust chamber and nozzle extensions. The turbopumps are sidemounted to obtain the maximum fixed nozzle area ratio within the envelope constraints and the feed system is highly integrated to minimize weight and hot gas ducting. The turbopumps feature hydraulic driven low speed pumps to minimize turbopump weight.

The engine has a 10:1 throttle capability within the thrust range of MR = 5.5 to MR = 6.5 which is achieved by gas/gas injector concept in both the preburner and main injector. All hydrogen is vaporized in the regeneratively cooled thrust chamber assembly. The main injector oxidizer flow is vaporized in the main injector oxidizer vanes by the fuel rich turbine exhaust gases. The preburner oxidizer is vaporized in the preburner chamber, permitting a light weight design. The throttle capability may be used to increase the engine life and thermal cycle capability. Full engine thrust is usually only required during the first burn, and the subsequent burns are insensitive to thrust level. The off-design data indicate that throttling will significantly help the life capability of the critical components, since turbine temperature, turbine speed, and chamber wall temperature gradient rapidly decrease.

To control the engine, both two valve and a three valve control systems were considered. Poppet type valves with metallic seals were selected for all valve applications because of their inherent superior sealing capabilities over a wide range of temperatures. Electrical actuation was selected for all of the valves. Electrical actuation is desirable for systems which do not require fast response (as in the case of the OOS) because of the ease in checkout, long storage life, low maintenance and high reliability.

The two valve control system utilizes a liquid oxygen preburner valve for thrust control and the liquid oxygen pump discharge valve for mixture ratio control. Throttling is achieved by controlling the preburner mixture ratio. A three valve control system was used in the final design. In this system, an additional turbine and preburner bypass control valve was incorporated, because it provided superior engine design and operational features. The [illegible] three valve control system also provides the engine with built-in [illegible] capability for both the pressure fed and pump assisted idle mode.

The engine off-design and start transient characteristics were evaluated using the LETS II engine design computer program to verify that the engine concept met the operational requirements. This evaluation also led to the definition of idle mode and start transient valve sequencing requirements. The engine system control interface consists of a separate engine computer-controller and data storage translating and controlling the vehicle commands. The engine system control modifications for different applications were defined for manned, unmanned reusable, and expendable mission requirements.

Engine maintainability was considered in the design of each component to meet the cycle life, duration, maintenance, and refurbishing cost goals. Components of similar life capability were interfaced to obtain replaceable units.

The result of the engine study indicated that use of the retractable nozzle concept for high pressure engines (such as the stage combustion engine cycle) result in very marginal payload gains. Conversely, a retractable nozzle improves the capability of the low pressure cycle engines considerably. Therefore, a fixed nozzle engine concept was selected for both the 10K and 25K thrust engine configuration.

The driving design consideration was the requirement of engine reusability and life, particularly in the thrust chamber, and turbine design area. These requirements were met by adequately controlling the thermal condition in the thrust chamber (and throughout the engine) and the selection of proper turbine tip speeds.

The study also indicated that the small thrust engine results in higher payload capability due to the feasibility of larger nozzle expansion area ratios within the given envelope constraints. However, the smaller thrust engines resulted in lower payload capability when clustered in a multi engine system as compared to a single engine system of equal thrust level.

The engine nominal design data are summarized for the 25K and 10K engine in Tables I and II. Figures 1 and 2 show the baseline engine configuration for the 25K and 10K engines respectively.

The results of the engine cost analysis indicates that the engine cycle selection and thrust level selection have very little impact on engine development and engine production cost. For the engine cost estimates, the schedule shown in Figure 3 was baselined resulting in the costs summarized in Table III.

In the course of the OOS Engine Design Study, it became evident that in certain technology areas more information is required.

The engine performance presented for 99% ERE is conservative because of the JANNAF performance prediction method used. A new JANNAF interim method results in 1-1/2 to 2 sec higher performance at an engine mixture ratio (MR)

TABLE I

25K ENGINE DESIGN SUMMARY

Cycle: Staged Combustion

Turbines: Parallel - Split Shaft (OTPA and FTPA)

Performance	
F, (lb)	25,000
MR	6.0
I_s (sec)	465.7
P_c, (psia)	1800
F/P_c	13.88
R_t (in.)	1.51
Envelope	
ϵ_o	290
ϵ_{trans}	N/A
Engine Overall Length (in.)	32.0
(L' = 6 in.) (L' Throat - Gimbal = 13.5 in.)	
Engine Stowed Length (in.)	N/A
Engine Exit dia (in.)	51.43
Engine Weight, W_{BO} (lb)	459.3

TABLE II

10K ENGINE DESIGN SUMMARY

Cycle: Staged Combustion

		Side Mounted Pumps Baseline
Performance		
F, (lb)		10,000
MR		6.0
I_s (sec)		465.5
P_c, (psia)		1250
F/P_c		8.0
R_t (in.)		1.13
Envelope		
ε_o		400
ε_{TRANS}		5.5:1
Engine Overall Length (in.)		76
(L' = 6 in.) (L' Throat - Gimbal = 13.0 in.)		
Engine Stowed Length (in.)		76
Engine Exit dia (in.)		45.20
Engine Weight, W_{BO} (lb)		278.1
Suction Lines		
Diameter	LO_2/Fuel	2.41/2.77
Location Diameter	LO_2/Fuel	16.0/16.8

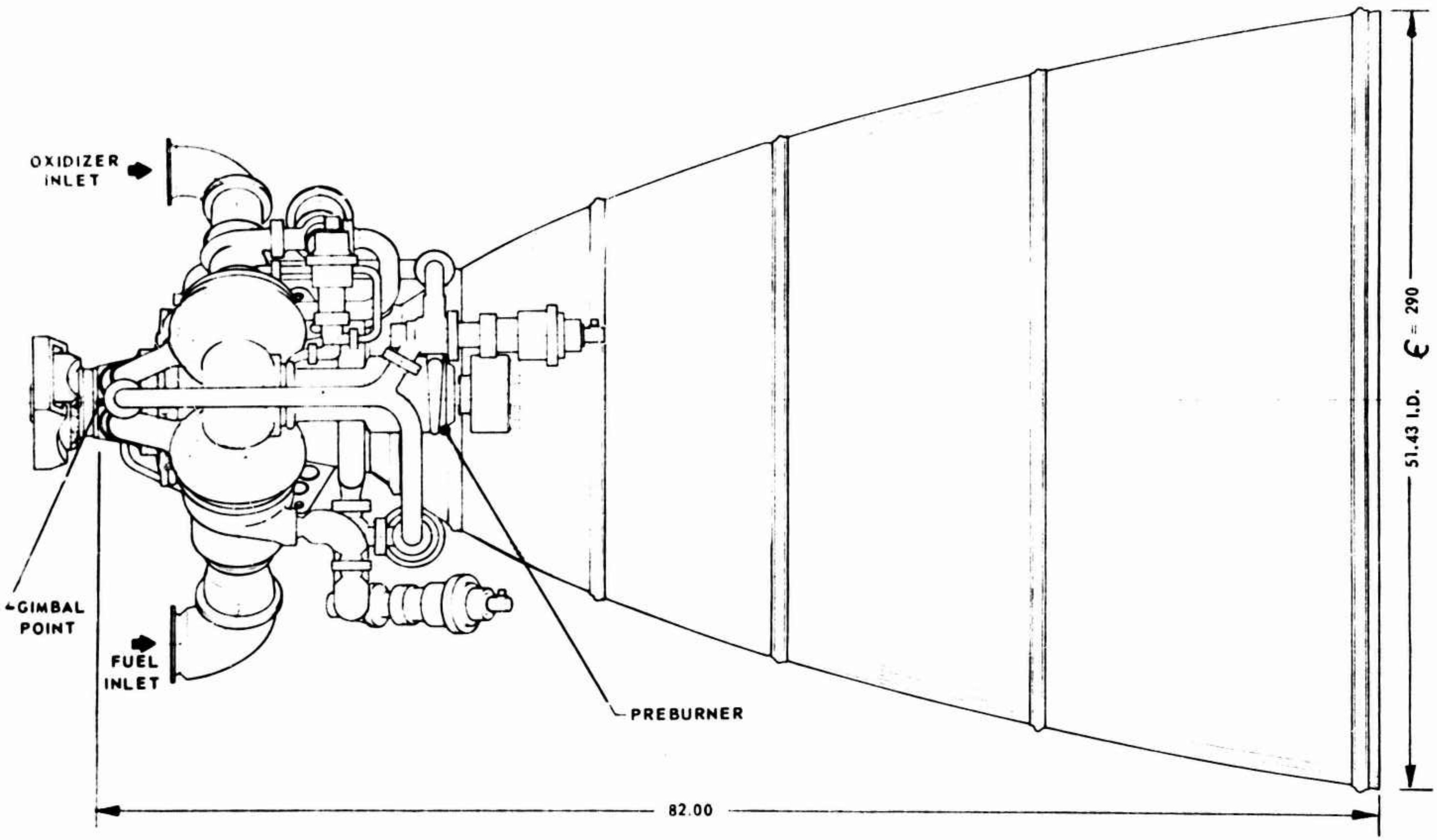


Figure 1. 25K Baseline Engine Configuration

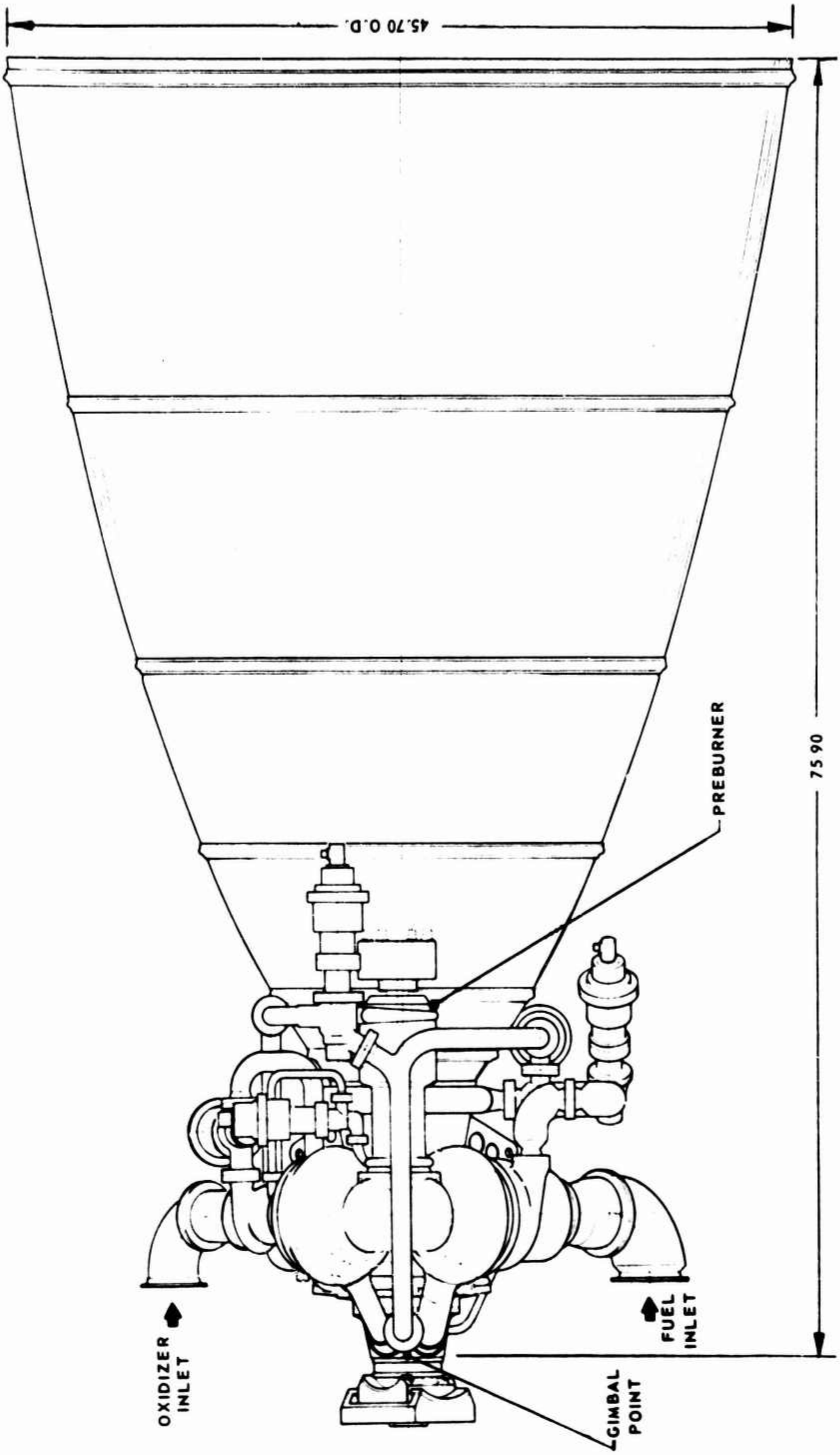


Figure 2. 10K Engine Baseline Engine Configuration

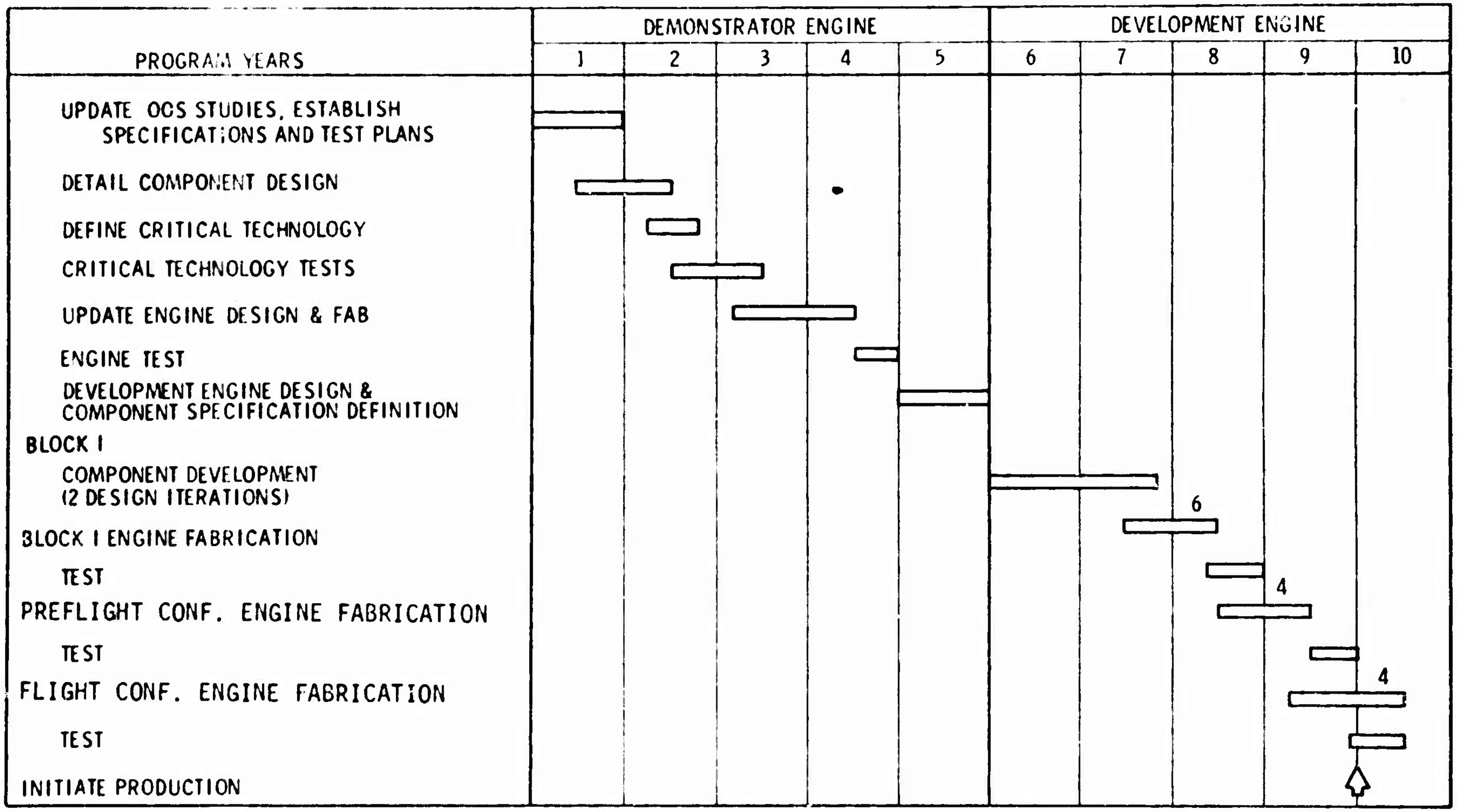


Figure 3. OOS Master Schedule

TABLE III

GROUND BASED REUSABLE 25K BASELINE ENGINE COST

DOLLARS

	Demonstrator Program	Development Program	Production Unit
Manufacturing	2,900,304	10,576,125	620,500
Quality Control	1,410,735	3,824,205	107,700
Test	6,407,000	15,194,000	40,600
Engineering	7,500,000	15,131,662	62,500
	18,218,039	44,725,992	831,300

of 6.0. Aerojet experience shows that at the higher chamber pressures 99.5% energy release efficiency (ERE) can be achieved. From this, it was concluded that the performance tolerance band is approximately 4 sec of specific impulse; the quoted performance presents the lower limit.

The high performance levels are obtained mostly through large nozzle expansion area ratios. (Much larger than the current TCA hot test experience covers). Therefore, it appears that large expansion ratio nozzles have to be investigated for their performance potential and deep throttling capability with LO_2/LH_2 propellant (ice formation).

Envelope restricted engine performance is strongly affected by the thrust chamber length. Consequently, injector concepts resulting in short chamber length are the most beneficial for this application.

The engine weight in the reusable mission has a large payload effect. The engine design analysis indicates that most of the engine weight is in the nozzle of engines featuring large nozzle area ratios. Careful selection of nozzle regenerative coolant tube thickness or alternate materials can effect the engine weight by as much as 20%. A large fraction of the engine weight consists of the control valves subsystems, so it appears a fruitful area for new technology is light weight valve and actuator development (Table IV).

The greatest need for more technology information was found to exist in the determination of the thrust chamber material. Low cycle fatigue life, in particular the effect of the operating environment, appears to have an order of magnitude effect on the low cycle fatigue life of the considered chamber materials. This technology is particularly important for low thrust engines, since thrust chamber thermal cycle life largely effects the thrust chamber pressure selection as well as the coolant pressure drop requirements, fuel pump discharge pressure, and hence the engine cycle power balance.

The pump fed low thrust engine appears to have some difficulties with component miniturization; turbopump assembly performances (TPA) suffer from large internal leakage losses, which adversely affect the power balance and the efficiency potential of multi-stage, high speed pumps. The TPA rolling contact bearings are very small, and they operate at high DN values. The result is that they must operate at very light loads, relying on good thrust balancing for long life. Bearing retainers must be of low profile to permit adequate cooling and very limited experience is available in this area.

Small reusable engines require a considerable amount of instrumentation to permit efficient maintenance. Current instrumentation technique and sensors contribute greatly to overall engine weight. Sensor miniturization would greatly help to reduce engine weight.

Engine maintenance requires additional flight readiness instrumentation data recording, and failure or performance degradation analysis capability. An engine mounted computer forming an integral engine component should be developed in parallel with the engine.

TABLE IV

25K BASELINE ENGINE WEIGHT SUMMARY

STAGED COMBUSTION - PARALLEL TURBINE DRIVE

F	= 25,000 LB	MR	= 6.0
ϵ_{EXIT}	= 290	NPSH (F)	= 60 FT NOM
P_c	= 1800 PSIA	NPSH (0)	= 16 FT

FIXED NOZZLE - TUBULAR - ALL H_2 REGENERATIVELY COOLED

	WEIGHT CATEGORY	WEIGHT - LB	% TOTAL WEIGHT
I.	TCA		
	A. INJECTOR	21.7	
	B. COPPER TC TO ϵ = 6:1	37.1	
	C. TUBE REGEN TO EXIT ϵ = 290:1	107.4	
	D. IGNITER (INCLUDES IGNITER VALVE)	12.1	
	SUBTOTAL	178.3	38.8
II.	TPA'S		
	A. FUEL (INCLUDING IN-LINE BOOST PUMP)	37.3	
	B. OXIDIZER (INCLUDING IN-LINE BOOST PUMP)	30.2	
	SUBTOTAL	67.5	14.7
III.	VALVES	43.1	9.4
IV.	GAS LINES, GAS MANIFOLDS, LIQUID LINES	78.3	17.0
V.	TURBINE DRIVE POWER - PREBURNER	24.7	5.4
VI.	GIMBAL ASSY AND ACTUATOR SUPPORT STRUCTURE	9.9	2.2
VII.	ENGINE HARNESS, INSTRUMENTATION, SUPPORT STRUCTURE, BRACKETRY, AND ATTACH HARDWARE	57.5	12.5
	TOTALS	459.3	100.0

NOTE: ENGINE CONTROLLER WEIGHT NOT INCLUDED.

Another maintenance requirement is that the engine be ground checked duplicating actual engine operation. This is very problematic for large area ratio nozzle engines and may require new nozzle technology, in particular, separable, light weight, radiation cooled nozzle extensions, which can be removed at low area ratio for sea level testing, without altering significantly the engine fuel system parameters, and without decreasing flight engine performance.

The operational requirements for multi-restart engines desire childown through idle mode. High pressure engine idle mode thrust levels are rather low. This is caused by the small throat size of these engines. Pump assisted idle mode, would double the idle mode thrust level. Pump assisted idle modes may present mixture ratio control problems due to two phase flow and should be developed in an experimental program.

Engine dynamic stability was also considered. Injector baffles and acoustic dampers were, therefore included in both the main and preburner injectors. Both of these concepts are relatively new technology and need extension.

The requirement of autogenous tank pressurization during pump assisted idle mode was a new operating requirement and requires demonstration.

Engine start transients were kept slow to minimize the thermal shock on the turbine disks and blades. Technology could be developed to lower these transient stresses by special disk designs permitting faster starts and improved impulse capability for short impulse operation.

SECTION III

TECHNICAL DISCUSSION

A. ENGINE DESIGN PARAMETRIC STUDY

This section of the final report contains the results of the parametric engine design study including:

- Generalized cycle descriptions in the range of 8K to 50K thrust level (excluding engine design constraints) for mixture ratio = 6.0.
- Comparison of the engine cycle payload capability considering envelope and engine design constraints at various thrust levels and MR = 6.0.
- Identification of the best engine cycle at each thrust level between 8K and 50K at MR = 6.0.
- Engine cycle description of the best engine cycle at discreet thrust levels of 8K, 15K, and 25K at MR = 6.0.

1. Summary of Requirements

The operating characteristics requirements for the engine parametric study are summarized in Table V and the stowed envelope constraints in Table VI. The vehicle payload sensitivity to engine specific impulse and engine weight are given for two missions both the Orbit-to-Orbit Mission and the Lunar Lander Mission and are summarized in Table VII.

2. Evaluation Criteria

The evaluation of the engine cycles consisted of three major elements.

- Payload Capability
- Operational Capability
- Engine Cost

A more detailed breakdown of these characteristics is shown in Table VIII. For the parametric analysis the engine payload characteristics were evaluated with respect to satisfying the operational capabilities. The engine cost impact due to thrust size and cycle selection was found to be insignificant and was not used as a cycle evaluation criterion. The selection of the best engine cycle for a given thrust is based on the payload capability only.

TABLE V

8,000 TO 50,000-POUND THRUST ENGINE OPERATING CHARACTERISTICS

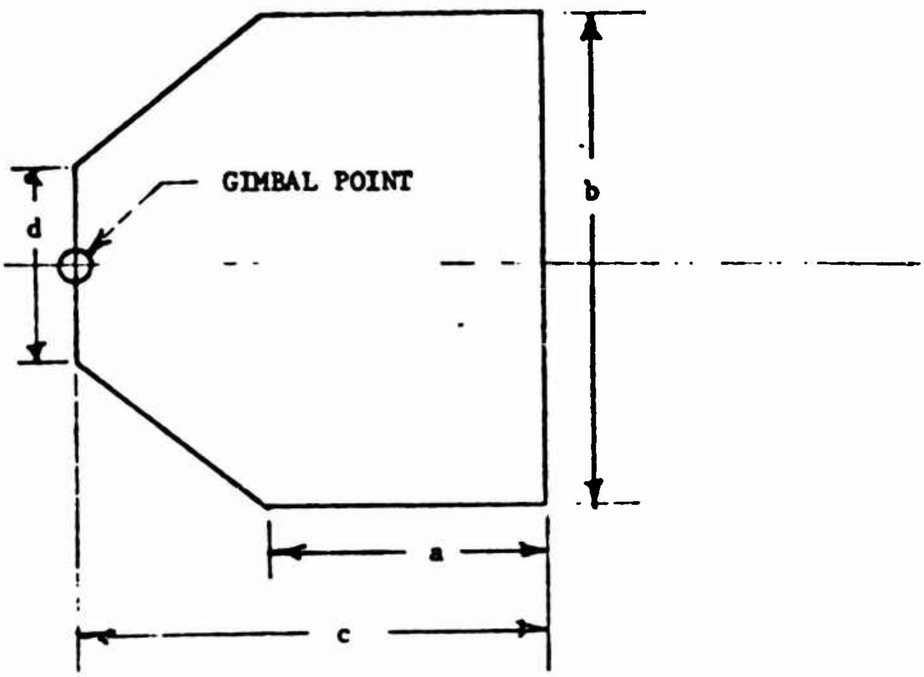
Propellants	Liquid Oxygen/Liquid Hydrogen
Maximum Vacuum Thrust, pounds	8,000 to 50,000
Nominal Engine Mixture Ratio	6.0:1
Engine Mixture Ratio Operating Range	5.5:1 to 6.5:1
Vacuum Thrust Throttling Capability	5.0:1
Nozzle Configuration	*
Nozzle Expansion Ratio	*
Turbine Drive Cycle	*
Vacuum Specific Impulse, seconds	*
Engine System Weight, pounds	*
Number of Vacuum Starts	60
Lifetime (Expendable Mode), thermal cycles	6
Lifetime (Reusable Mode), thermal cycles	300
Lifetime (Reusable Mode), hours	10
Gimbal Angle (Square Pattern), degrees	7
Gimbal Acceleration, radians/(second)^2	20
Minimum Natural Frequency of Gimbal System, Hertz	10
Fuel Pump NPSH, feet of hydrogen	60
Oxidizer Pump NPSH, feet of oxygen	16
Maximum Single Run Duration, seconds	1000
Maximum Storage Time in Orbit (Dry), weeks	52
Maximum Time Between Firings (Coast Time), days	14
Minimum Time Between Firings (Coast Time), minutes	10
Maintenance-Free Engine Run Time, hours	2
Maintenance-Free Engine Firing Cycles	60

*To Be Determined as a result of design and analysis.

TABLE VI

MAXIMUM ENGINE DIMENSIONS (STOWED POSITION IN COS)

DIMENSION	MAXIMUM VALUE
a	50 inches
b	87 inches
c	82 inches
d	40 inches



ENGINE CROSS-SECTION THROUGH THRUST CENTERLINE

TABLE VII

VEHICLE TRADEOFF FACTORS

Tradeoff factors based on Orbit-to-Orbit Missions:

$$\frac{\Delta PL}{\Delta Isp} = 157 \text{ lb/sec}$$

$$\frac{\Delta PL}{\Delta W_{Burnout}} = -3.68$$

Tradeoff factors for Lunar Lander (Based on 20,000 lbs to and return)

$$\frac{\Delta PL}{\Delta Isp} = 114 \text{ lb/sec}$$

$$\frac{\Delta PL}{\Delta W_{Burnout}} = -1$$

TABLE VIII

METHOD OF CYCLE EVALUATION

Payload Capability	Operational Capability	Cost
Specific Impulse	Throttling	Demonstrator Engine
Engine Weight	Chill Down	Flight Engine Dev.
Mixture Ratio	Propellant Utilization	Production
Envelope	Autogenous Pressurization	Facilities
NPSH	Engine Control (SS)	
	Transient Control	
	Engine Life	

- Engine Payload Capability Primary OOS Engine Cycle Selection Criteria

3. Engine Cycle Description

The basic engine cycles included the following six cycles (see Figures 4 and 5).

a. Bleed Cycles

The bleed cycles shown schematically in Figure 4, feature a series turbine drive, and for illustration, a separate turbine exhaust nozzle. These engine cycles are degraded in performance due to the turbine exhaust losses, and performance losses due to TCA mixture ratio shift induced by the propellant bleed. The bleed cycles usually are operated at MR = 5.0 to minimize these losses. At the mixture ratio 6.0 these losses increase. Consequently, bleed cycles are not competitive with the topping cycles.

The practical thrust chamber operating pressure of the bleed cycles is dependent on the TCA chamber life requirements and the available engine envelope.

Some basic assumptions were made in the evaluation of the bleed cycles. The turbine temperature of the gas generator cycles was assumed to be 1860°R for all thrust sizes and chamber pressures. For the chamber tap-off bleed cycle, the same turbine temperature of 1860°R was assumed at comparable mixture ratio as for the gas generator cycle.

For the coolant bleed cycle, the turbine temperature of 1000°R was assumed for all thrust sizes and chamber pressures.

b. Topping Engine Cycles

These cycles have high performance yielding engine performance approaching the thrust chamber performance because bleed losses are nonexistent or minimized. The basic schematic of these cycles is shown in Figure 5.

The expander cycle is inherently a low pressure cycle since the available energy for the feed system drive is the latent heat from the cooling jacket. The TCA pressure level attainable is strongly dependent on the thrust size and assumed thrust chamber length. It is also limited by the power balance requirement of the TPA feed system.

The staged combustion cycle have a fuel-rich preburner yielding a cycle somewhat more complex than the expander cycle. However, this addition permits the engine to operate at high chamber pressures and also makes the cycle less sensitive to feed system component efficiency. The turbine temperature was restricted to 1860°R for all thrust sizes and chamber pressures.

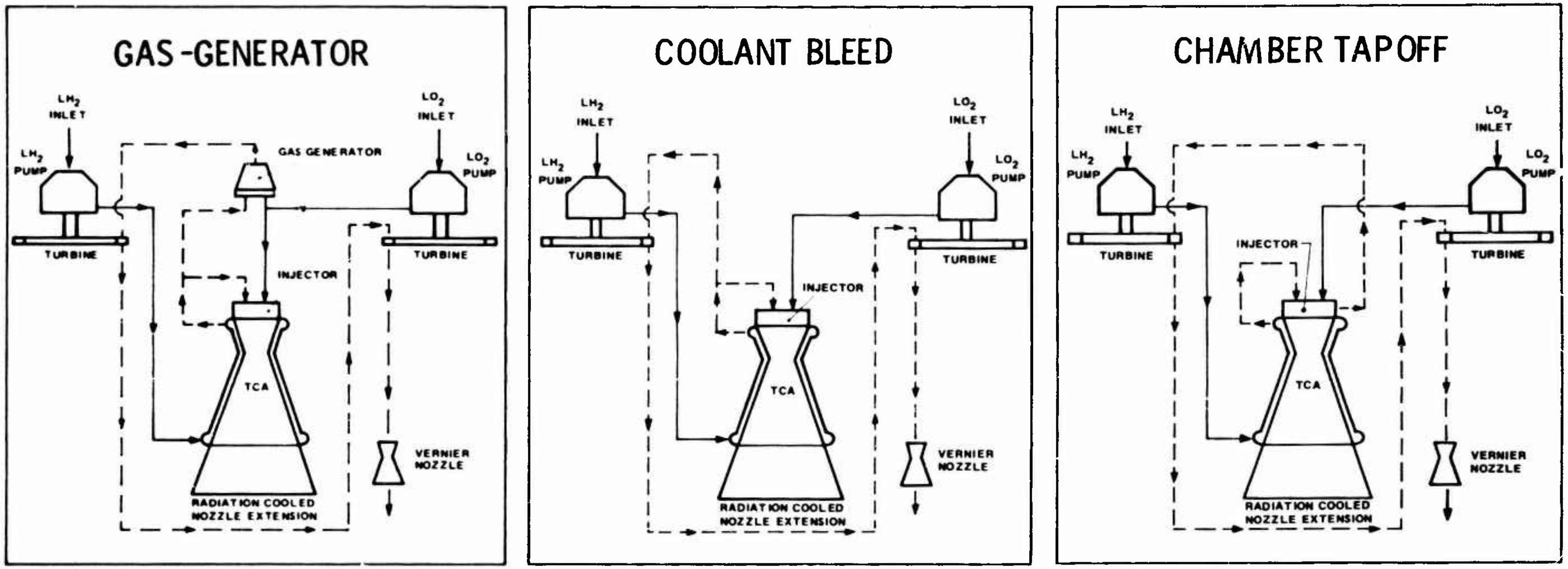


Figure 4. Engine Cycle Description - Bleed Cycles

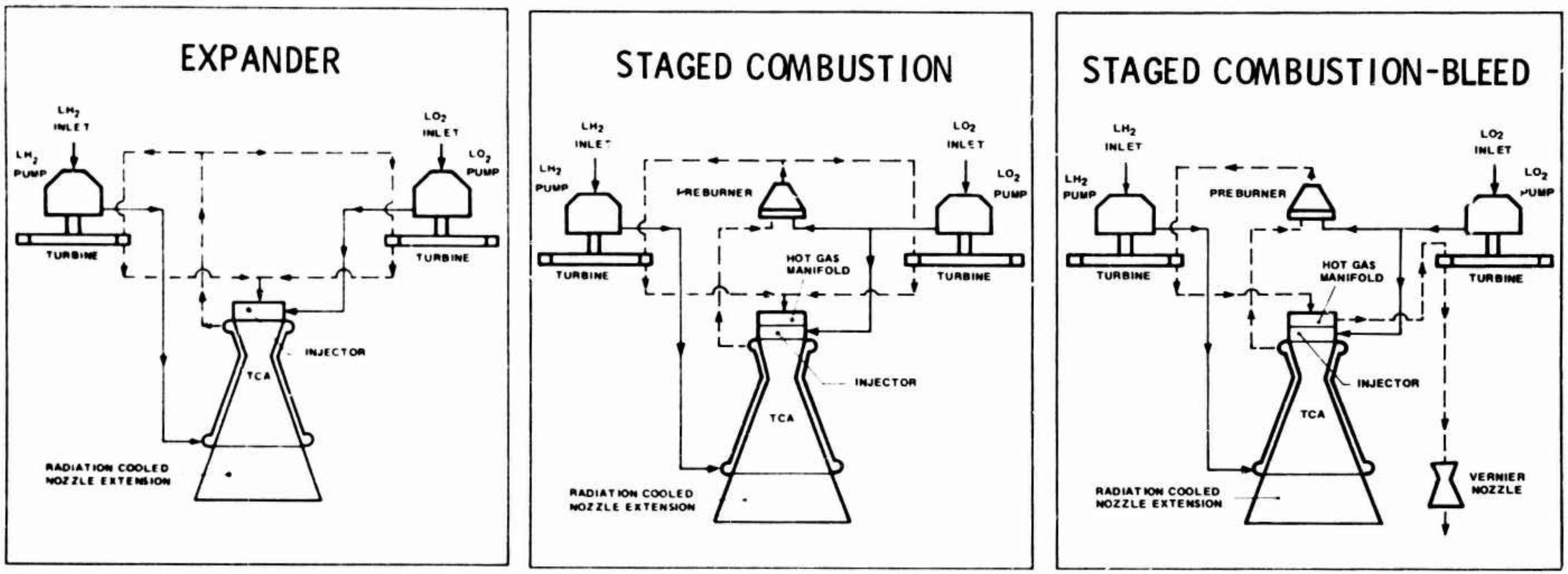


Figure 5. Engine Cycle Description - Topping Cycles

The staged combustion bleed cycle is a slight modification of the basic staged combustion cycle and has its application where power balance limits are encountered and bleed flow is desirable (i.e., dump-cooled nozzle extensions). All topping cycle feed systems are driven by parallel turbine drives to permit deep throttling without reverse pressure gradients at the tip of the oxidizer turbine, as would be the case for series turbine drives.

4. Method of Approach

The general description of the six basic engine cycles at mixture ratio 6.0 and thrust levels between 8K and 50K involved the definition of the following parameters:

Engine Delivered Vacuum Specific Impulse

Engine Weight

Engine Length

Engine Maximum Diameter

Suction Line Diameter

Suction Line Orientation

All engines were designed to meet the 300 engine thermal cycle requirements, which in turn defined the coolant pressure drop requirements and bulk temperature rise. All engines were designed to operate at a nominal mixture ratio of 6.0, a mixture ratio range of 5.5 to 6.5, and a throttle capability of 5:1.

Additional analysis conducted to establish the engine thermal conditioning requirements and program cost is discussed in separate sections.

a. Thrust Chamber and Engine Performance

The contract specified that the engine delivered specific impulse be defined by the JANNAF method. A simplified performance summary is shown in Figure 6, describing TCA performance as a function of chamber pressure, expansion area ratio, mixture ratio and thrust level. There are currently two JANNAF performance prediction methods used. The interim method yields about 1.5 sec higher performance at MR = 6.0 and considerably higher performance at the larger mixture ratios. For this study the official JANNAF performance prediction method was used yielding lower performance than the newer interim method. A detailed description of the engine performance is presented in Appendix A including both sets of data.

Engine performance was evaluated by utilizing existing computer programs for all six engine cycles. For this purpose thrust chamber cooling requirements and engine control systems were defined to establish feed system pressure drop requirements. A first set of cooling requirements were assumed for constant coolant Mach No. of the TCA throat Ma = 0.45. For the

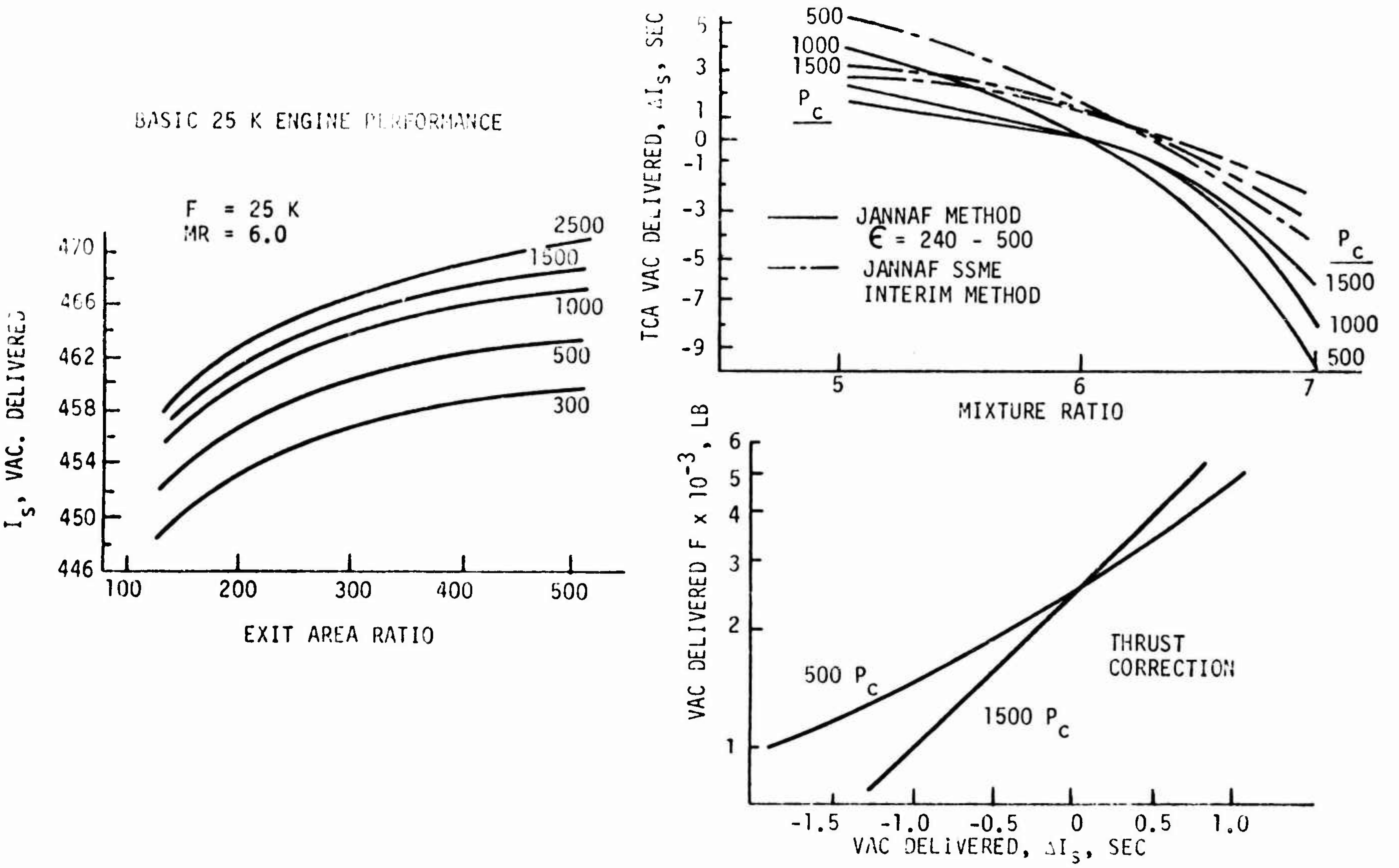


Figure 5. TCA Performance For Power Balance Analysis

staged combustion cycles and the expander cycle, the cooling requirements were later revised to adjust the coolant pressure drop requirement to meet the thrust chamber life requirements and the power balance simultaneous, for each thrust level and chamber pressure. This modification showed little effect on the staged combustion cycle.

A further basic assumption for the performance evaluation was made in the selection of the nozzle contour. As the basic contour for all thrust levels, chamber pressure, and mixture ratios, the minimum length Rao contour was assumed as the optimum contour. The injector energy release efficiency was assumed to be ERE = 99% by agreement with the customer. ALRC experience has shown this assumption is well within the existing state-of-the-art.

b. Engine Weight and Envelope

No standard methods are in existence to determine engine weights as a function of engine cycle, thrust level, and chamber pressure. Consequently, a new procedure had to be developed.

The first step in this procedure was to define common technology, design, and materials, for all engine cycles. All assumptions made for this purpose are documented in Section III,A,5 and are summarized in Figure 7.

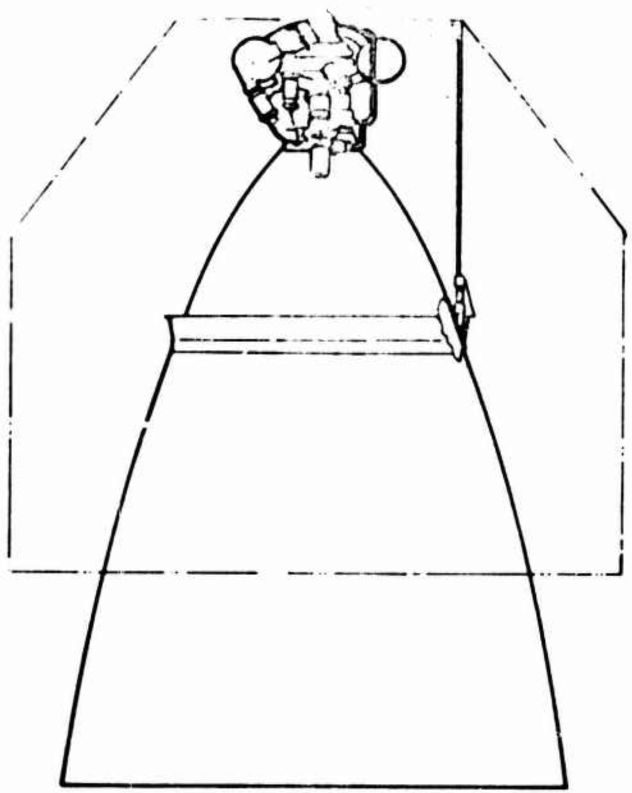
Both fixed and retractable bell nozzle engine configurations were considered. For the retractable nozzle concept certain groundrules were imposed to define the manner of retracting the nozzle extension. As shown in Figure 8, both minimum weight and length nozzles were investigated.

The minimum weight concept retracts the nozzle just enough to clear the 82-in. interface and has the transition area ratio at the minimum area ratio. The minimum length retractable nozzle retracts the nozzle in such a manner as to obtain the minimum stowed engine length. The decision was made to consider the minimum weight retractable nozzle concept only, since no payload tradeoff factors were available to investigate the desirability or benefits of the reduced stowed engine length.

Based on these groundrules the procedure for the engine weight analysis was as follows:

- For each engine cycle, a baseline engine layout was prepared indicating the most desirable engine configuration at design, thrust, and chamber pressure.

- For each cycle every component was scaled analytically as functions of:



CHAMBER	
MATERIAL:	ZrCu
COOLING:	SINGLE UP-PASS
L':	8 INCHES (14 INCHES)
CONFIG:	MILLED SLOTS
T_W MAX:	1000° F
NOZZLE	
MATERIAL:	ARMCO 22-13-5
TYPE:	BRAZED TUBE
COOLING:	DOWN & UP-PASS
ATTACHMENT:	BRAZED
T_W MAX:	1200° F
CONTOUR	MIN LENGTH RAO
SKIRT:	
MATERIAL:	AGCarb-101
TYPE:	RADIATION COOLED
ATTACHMENT:	FLANGE & SEAL

FIXED & RETRACTABLE

MAIN INJECTOR	
PROPELLANT:	GH_2/LO_2
TYPE:	PLATELETS - VANE
MATERIAL:	ARMCO 22-13-5 & CRES 347
ERE	99%
GAS GENERATOR & PREBURNER	
PROPELLANT:	GH_2/LO_2
TYPE:	PLATELET, DUAL OX CIRCUIT
MATERIAL:	INCONEL 718
T_{Ti} =	1860°R
TURBOPUMPS	
DRIVE INDIVIDUAL TURBINES	
BOOST PUMPS:	MIN TPA WEIGHT

Figure 7. Main Assumptions for Parametric Study

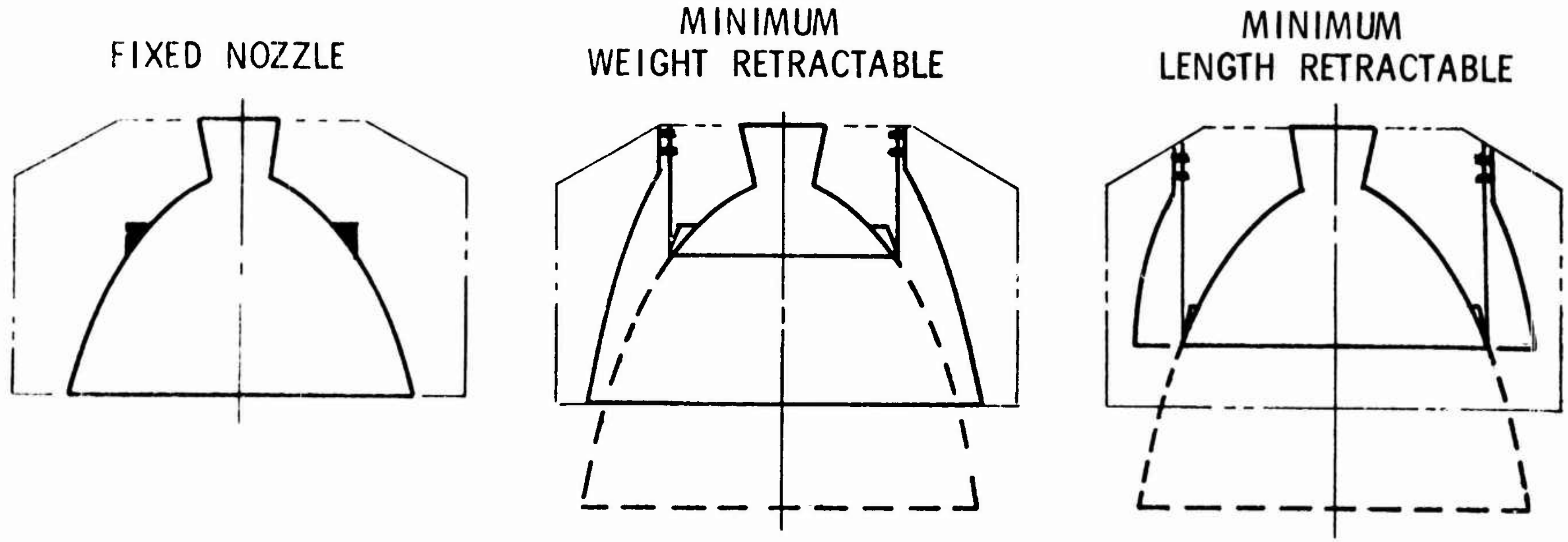


Figure 3. Nozzle Configuration

Thrust

Chamber Pressure

Mixture Ratio

Expansion Area Ratio

NPSH

- The engine weights were computed as a function of the scaling parameter for each engine cycle and nozzle concept.

For the selected baseline design points the component weights were calculated based on the engine layout drawings and are presented in Table IX through Table XIV. These weights form the base for the analytical weight allocation.

c. Engine Envelope Dimensions

The engine envelope dimensions are presented as the study total engine length and the maximum nozzle extension diameter. It was assumed that the engine length from the chamber throat to the gimbal point is constant for all cycles, except the expander cycle, and for all thrust levels and chamber pressures.

For the expander cycle a thrust chamber length of 14-in. was assumed as compared to 8-in. for all other cycles. This improves the drive capability of the cycle since it will result in increased coolant temperatures. Consequently, engine length variations are equal to the minimum length Rao Contour Nozzle length variation.

d. Suction Line Diameter and Orientation

The suction line orientation was scaled from the engine layout drawings for each engine cycle. The orientation is expressed in terms of suction line center diameter from the thrust chamber axis.

e. Thrust Chamber Cooling Requirements

For the evaluation of the engine performance and weight, it is necessary to establish the thrust chamber cooling requirement since it effects the turbine bleed losses, power balance capability, and pump discharge requirement.

The objective of the thrust chamber cooling system is to provide a reusable chamber capable of 300 thermal cycles. For this purpose a zirconium copper regenerative cooled chamber with machined cooling passages was selected.

TABLE IX

WEIGHT BREAKDOWN

SUBJECT ENGINE CYCLE — GAS GENERATOR BLEED - SERIES TURBINE DRIVE

F = 10,000 LBS. MR = 6.0

ϵ_{EXIT} = 450 ϵ_{TRANS} = 150

P_c = 1250 PSIA NPSH(F) = 60 FT. NOMINAL
NPSH(O) = 16 FT.

FIXED AG CARB Radiation Cooled Nozzle Extension

NOTE: Engine Controller Unit not included

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
I. TCA		
A. Injector	24.0	
B. Copper TC to ϵ = 6:1	29.5	
C. Tube Regen to ϵ = 150:1	17.4	
D. Rad. Cooled Nozzle to ϵ = 450:1 + Fixed Flange	39.7 8.0	
E. Igniter	8.4	
F. Including Nozzle Retraction (None)	127.0	43.0%
II. TPA's		
A. Fuel	17.0	
B. Ox	24.0	
C. Including Boost Pumps (None)	41.0	13.9%
III. Valves	50.2	16.9%
IV. Gas Lines, Gas Manifolds, Liquid Lines	18.2	6.2%
V. Turbine Drive Power - (GG & Igniter)	20.6	6.9%
[illegible] Harness, Instrumentation [illegible] Structure, Bracketry [illegible] Hardware	38.6	13.1%
	295.6	100.0%

TABLE X

WEIGHT BREAKDOWN

[illegible] ENGINE CYCLE COOLANT TAPOFF - SERIES TURBINE

F = 10,000 LBS. MR = 6.0

ϵEXIT = 450 ϵ_{TRANS} = 150

P_c = 1250 PSIA NPSH(F) = 60 FT. NOMINAL

NPSH(O) = 16 FT.

FIXED AG CARB Radiation Cooled Nozzle Extension

NOTE: Engine Controller unit not included

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
1. TCA		
A. Injector	24.0	
B. Copper TC to ϵ = 6:1	29.5	
C. Tube Regen to ϵ = 150:1	17.4	
D. Rad. Cooled Nozzle to ϵ = 450:1 + Fixed Flange	39.7 8.0	
E. Igniter	8.4	
[illegible]. Including Nozzle Retraction (None)	127.0	46.7%
[illegible] TPA's		
A. Fuel	17.0	
[illegible]. Ox	24.0	
C. Including Boost Pumps (None)	41.0	15.1%
[illegible]	46.9	17.2%
[illegible], Gas Manifolds, [illegible]	18.6	6.8%
[illegible] Drive Power - (TCA)	-	-
[illegible], Instrumentation [illegible], Bracketry [illegible]	38.6	14.2%
	272.1	100.0%

TABLE XI

WEIGHT BREAKDOWN

SUBJECT ENGINE CYCLE — CHAMBER TAPOFF - SERIES TURBINE

$F =$ 10,000 lbs — MR = 6.0

$\varepsilon_{EXIT} =$ 450 — $\varepsilon_{TRANS} = 150$

$P_c =$ 1250 PSIA — NPSH(F) = 60 FT. NOMINAL
NPSH(O) = 16 FT.

FIXED AG CARB Radiation Cooled Nozzle Extension

NOTE: ENGINE CONTROLLER UNIT NOT INCLUDED

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
I. TCA		
A. Injector	24.0	
B. Copper TC to $\varepsilon = 6:1$	29.5	
C. Tube Regen to $\varepsilon = 150:1$	17.4	
D. Rad. Cooled Nozzle to $\varepsilon = 450:1$ + Fixed Flange	39.7 8.0	
E. Igniter	8.4	
F. Including Nozzle Retraction (None)	127.0	46.7%
II. TPA's		
A. Fuel	17.0	
B. Ox	24.0	
C. Including Boost Pumps (None)	41.0	15.1%
III. Valves	46.9	17.2%
IV. Gas Lines, Gas Manifolds, Liquid Lines	18.6	6.8%
V. Turbine Drive Power -(TCA)	-	-
[illegible] Harness, Instrumentation [illegible] Structure, Bracketry [illegible] Hardware	38.6	14.2%
[illegible]	272.1	100.0%

TABLE XII

WEIGHT BREAKDOWN - EXPANDER

SELECT ENGINE CYCLE EXPANDER - PARALLEL TURBINE DRIVE

F = 10000 lbs. MR = 6.0

ε_{EXIT} = 450 ε_{TRANS} = 145

P_c = 600 PSIA NPSH(F) = 60 FT. NOM.
NPSH(O) = 16 FT.

RETRACTABLE AG CARB Radiation Cooled Nozzle Extension

NOTE: ENGINE CONTROLLER UNIT NOT INCLUDED

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
I. TCA		
A. Injector	17.8	
B. Copper TC to ε = 6:1	40.5	
C. Tube Regen to ε = 145:1	40.4	
D. Rad. Cooled Nozzle to ε = 450:1	83.9	
E. Igniter	8.4	
F. Including Nozzle Retraction	71.1 + 191 = 262.1	62.8%
II. TPA's		
A. Fuel	17.0	
B. Ox	24.0	
C. Including Boost Pumps (none)	41.0	9.8%
III. Valves	47.3	11.3%
IV. Gas Lines, Gas Manifolds, Liquid Lines	28.1	6.8%
V. Turbine Drive Power - (TCA)	-	
[illegible] s, Instrumentation [illegible] ture, Bracketry [illegible]	38.6	9.3%
	417.1	100.0%

TABLE XIII

WEIGHT BREAKDOWN

SUBJECT ENGINE CYCLE STAGED COMBUSTION - PARALLEL TURBINE DRIVE

F = 25000 lbs. MR = 6.0

εEXIT = 450 ε_{TRANS} = 145

P_c = 1500 PSIA NPSH(F) = 60 FT. NOM.

NPSH(O) = 16 FT.

RETRACTABLE AG CARB Radiation Cooled Nozzle Extension

NOTE: ENGINE CONTROLLER UNIT NOT INCLUDED

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
I. TCA		
A. Injector	17.8	
B. Copper TC to ε = 6:1	33.3	
C. Tube Regen to ε = 145	40.4	
D. Rad. Cooled Nozzle to ε = 450:1	83.9	
E. Igniter	8.4	
F. Including Nozzle Retraction	70.6+183.8 = 254.4	53.6%
II. TPA's		
A. Fuel	36.0	
B. Ox	20.0	
C. Including Boost Pumps	20 + 56.0 = 76.0	16.0%
III. Valves	48.1	10.1%
IV. Gas Lines, Gas Manifolds, Liquid Lines	39.8	8.4%
V. Turbine Drive Power - Preburner (Includes Igniter)	18.1	3.8%
[illegible] Harness, Instrumentation [illegible] Bracketry [illegible]	38.6	8.1%
[illegible]	475.0	100.0%

TABLE XIV

WEIGHT BREAKDOWN

SUBJECT ENGINE CYCLE — STAGED COMBUSTION - BLEED SERIES TURBINE DRIVE

F = 25000 LBS. MR = 6.0

ϵEXIT = 450 ϵ_{TRANS} = 145

P_c = 1500 PSIA NPSH(F) = 60 FT. NOM.

NPSH(O) = 16 FT.

RETRACTABLE AG CARB Radiation Cooled Nozzle Extension

NOTE: ENGINE CONTROLLER UNIT NOT INCLUDED

WEIGHT CATEGORY	WEIGHT- LBS.	% TOTAL WEIGHT
I. TCA		
A. Injector	17.8	
B. Copper TC to ϵ = 6:1	33.3	
C. Tube Regen to ϵ = 145:1	40.4	
D. Rad. Cooled Nozzle to ϵ = 450:1	83.9	
E. Igniter	8.4	
F. Including Nozzle Retraction	70.6+183.8 = 254.4	51.4%
II. TPA's		
A. Fuel	36.0	
B. Ox	20.0	
C. Including Boost Pumps	20.0+56.0 = 76.0	15.4%
[illegible]. Valves	63.7	12.8%
[illegible]. Ox Lines, Gas Manifolds, Liquid Lines	41.7	8.4%
[illegible]. [illegible]ine Drive Power - Preburner (includes Igniter)	20.6	4.2%
[illegible] Harness, Instrumentation [illegible] Structure, Bracketry [illegible] Hardware	38.6	7.8%
[illegible]	495.0	100.0%

The requirement for 300 thermal cycles imposed severe design restraints on the thrust chamber because all temperature gradients must be maintained below certain limits.

Aerojet test data on low cycle fatigue characteristics of Zirconium Copper was used as bases for these limits. This data lead to the design requirement of a maximum wall temperature of 935°F. See Figure 9.

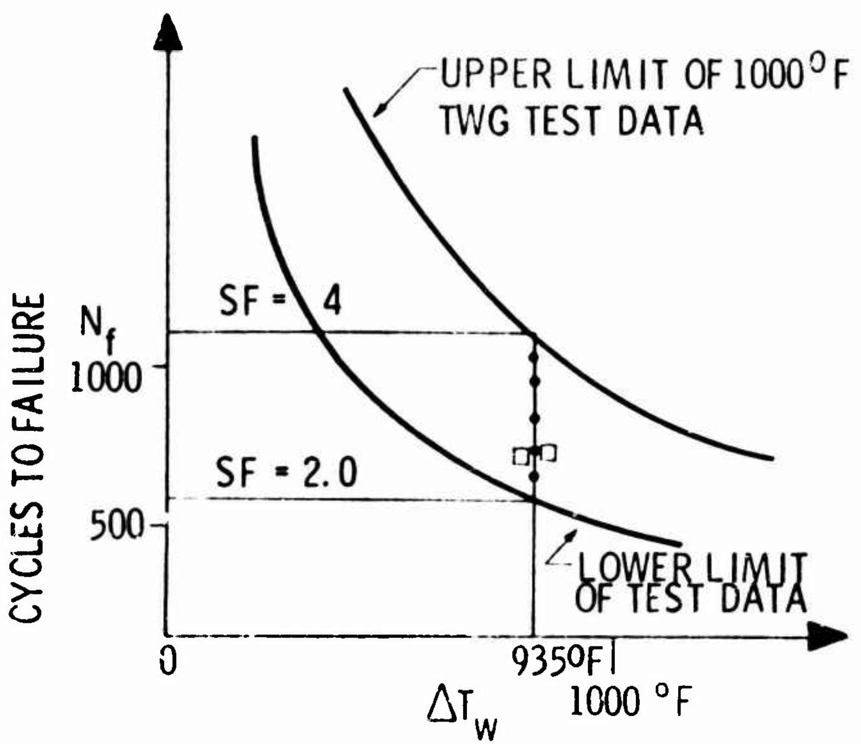
Figures 10 and 11 present the coolant bulk temperature rise and coolant pressure drop as a function of chamber pressure, thrust size and pump discharge pressure. Correction curves are shown for varying combustion chamber lengths and mixture ratios.

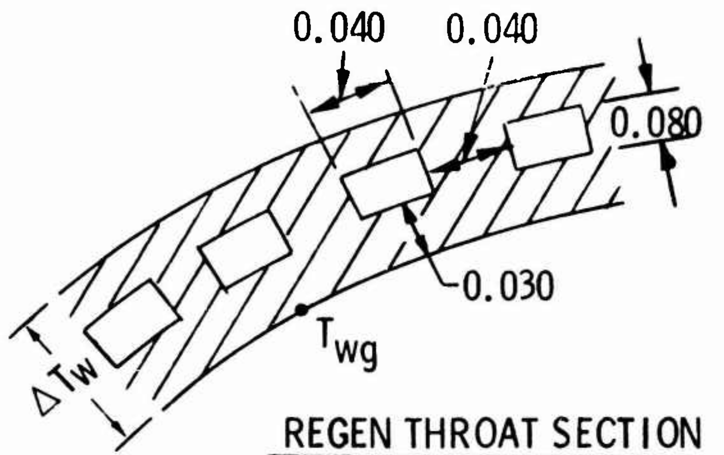
These coolant requirements were used for the generalized set of data. However, for the payload comparison analysis in which engine design constraints are important the coolant requirements were based on the heat transfer analysis (Appendix B) to meet 300 thermal cycles.

5. Assumptions

A complete listing of the engine technology and design assumptions made for parametric engine design study (Task IV) is included in Table XV which is self explanatory.

THRUST CHAMBER LOW CYCLE FATIGUE LIFE ΔT_w AT THROAT (ZrCu)





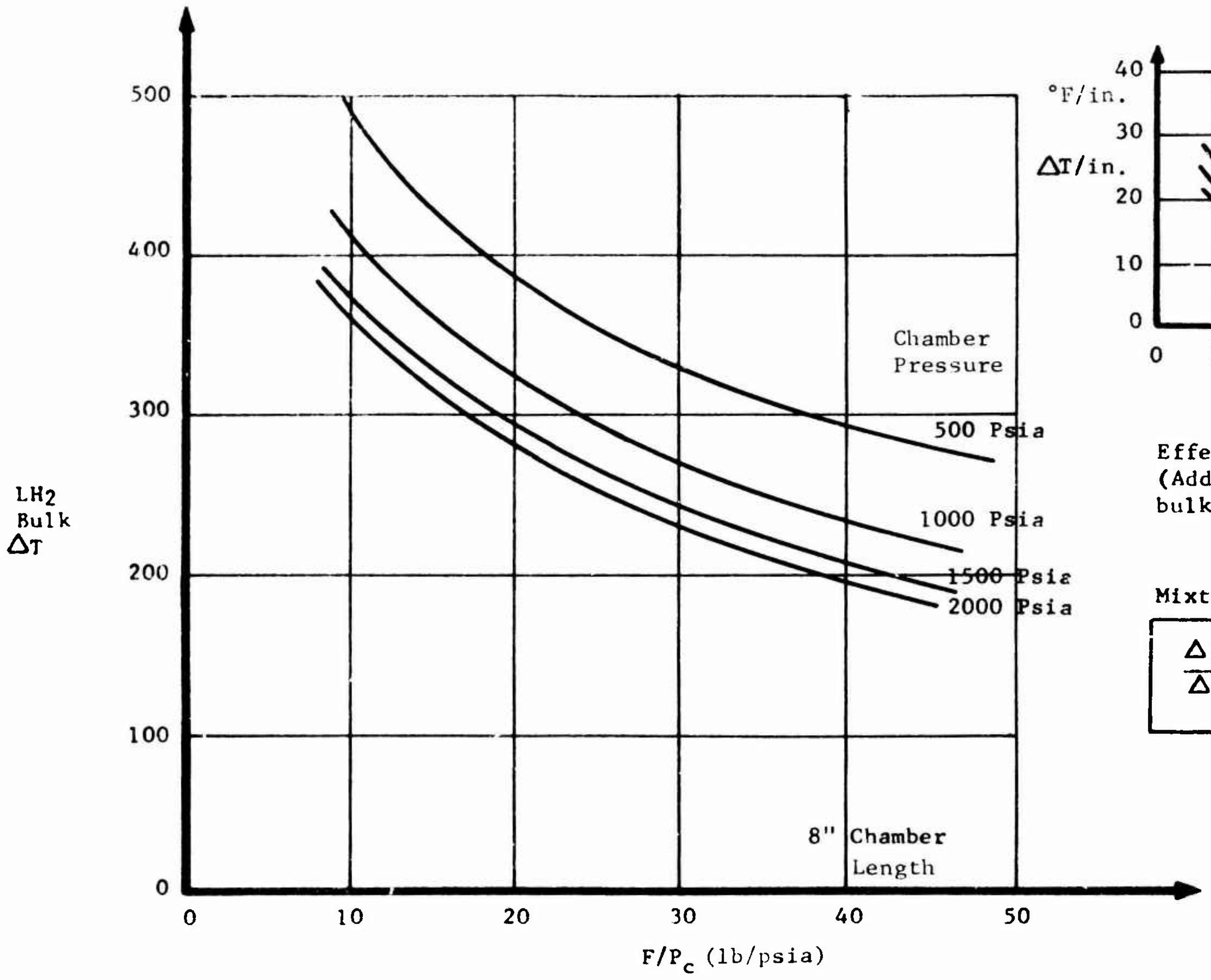
REGEN THROAT SECTION

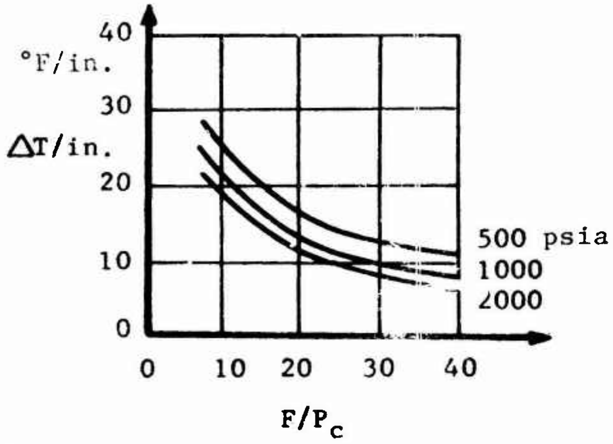
- DESIGN SAFETY FACTOR ON FATIGUE LIFE
- SF = 4.0 ON THEORETICAL CURVE
- SF = 2.0 ON LOWER DATA BOUNDARY

- FOR ZrCu CHAMBER AND N_f = 300 A THROAT ΔT_w = 935°F IS REQUIRED
- COOLANT MACH NO. MUST MEET THE REQUIREMENT OF ΔT_w = 935°F
- CHAMBER PRESSURES MUST BE LIMITED TO MEET ΔT_w = 935°F

  T_w = 1000°F

Figure). Chamber Cooling Requirement





Effect of Chamber Length
(Additive to 8-in. L' LH_2
bulk ΔT)

Mixture Ratio Correction

$$\frac{\Delta T_{MR}}{\Delta T_{MR = 6.0}} = \frac{MR + 1}{7}$$

Figure 10. Chamber Coolant Bulk Temperature Rise

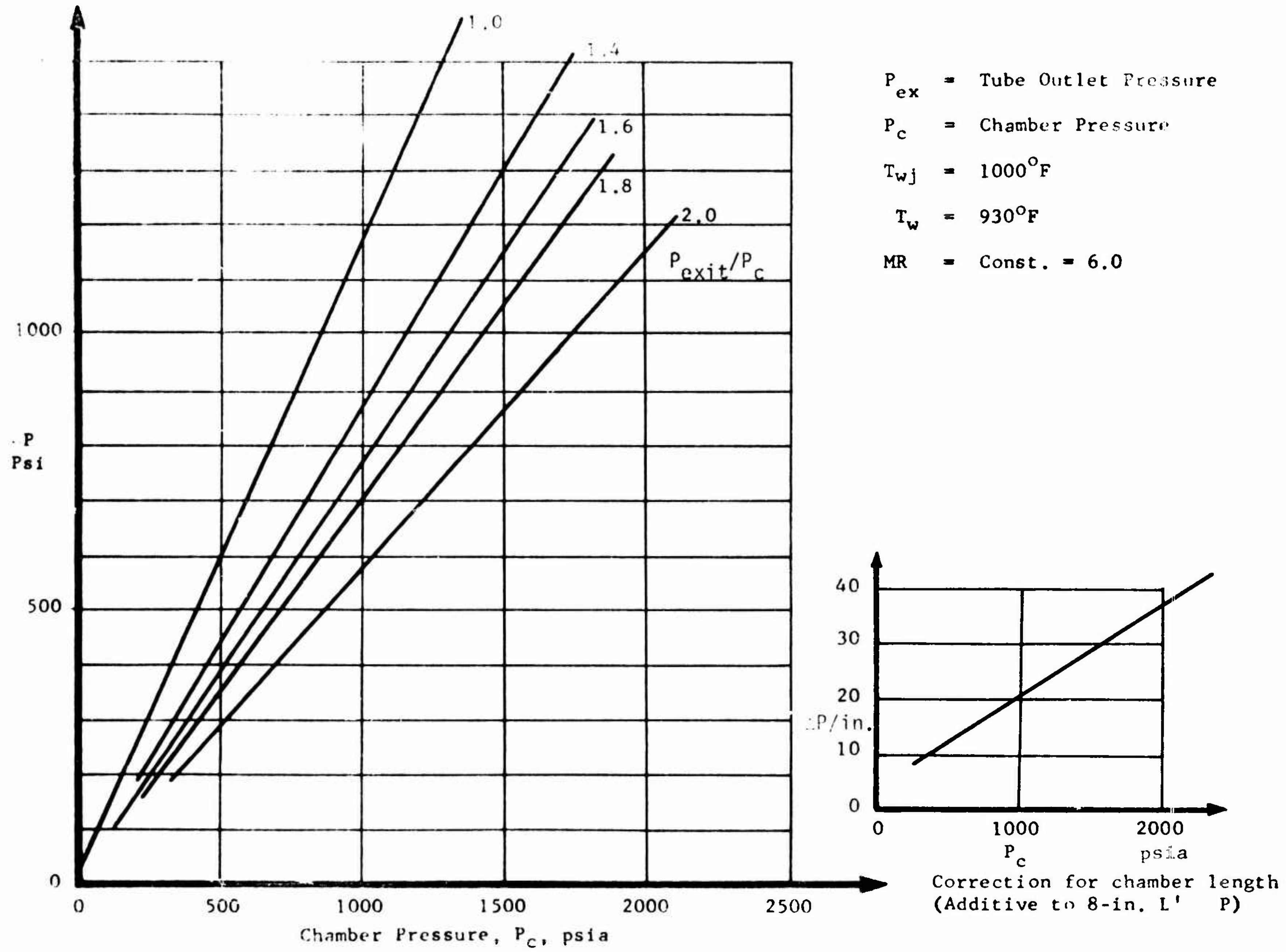


Figure 11. Chamber Coolant Pressure Drop

TABLE XV

OOS COMPONENT ASSUMPTIONS AND JUSTIFICATION

I. CHAMBER

	ITEM	ASSUMPTIONS	JUSTIFICATIONS
1.	Type	H_2 Regen. - Single "Up" Pass, Rectangular Channels	Best Coolant Properties; Less Expensiv for Less than 10 Units than Tubular Design
2.	Material	Z_rCu	Good Low Cycle Fatigue Prop., High Heat Flux Capability, Good LCF Data
3.	Configuration	Conical, with Short Cylindrical Section @ Injector End, & Long Cylindrical Section for Expander Cycle	Least Wt, Good Perf., Low Heat Transfe Area (Except for Expander Cycle)
4.	Film Cooling	None	Good Performance, Compatible Small F/E
5.	T Wall Max	1000°F	Good Low Cycle Fatigue and Strength Pr
6.	L' Chamber Length	8.0 in. (Except Expander Cycle)	Good Perf. Est. for Gas/Liquid Injecto
7.	Expander Cycle Chamber	Add Cylindrical Section	Need HOT H_2 to Run Cycle (See Graph fo ΔP/in. & ΔT/in. Estimates)
8.	Contraction Ratio	$(R_j/R_t)^2 = 2.0$ ($\Delta P_{ch} = 3.0\%$ for S.C. Cycles & 5.0% for non-S.C. Cycles)	Max. Performance in Short L', Min. Heat Transfer Area and Chamber Weight
9.	Chamber Area Ratio	($R_{exit}/R_t = \varepsilon_1$ = (See Chart) (Minimize to Reduce Weight)	See "Nozzle" Justifications
10.	Channel Configuration	Axial Milled Slots, Stepped Width	State-of-the-Art Fabrication; Smooth Ir Wall, Several Closure Techniques, Optim Cost vs ΔP and ΔWeight
	a. Land Width/Slot Width Ratio @ Throat	1.0 to 1.1	Per E-D Nozzle Proposal LR711618 and O Chamber Fab & Testing
	b. Slot Width, in. @	.040 in.	Per E-D Nozzle Proposal LR711618 and O Chamber Fab & Testing

TABLE XV (cont.)

ITEM		ASSUMPTIONS	JUSTIFICATIONS
	c. Land Width @ Throat:	.040 in.	Per E-D Nozzle Proposal LR711618 and O Chamber Fab & Testing
	d. Channel Depth	As Required for Cooling (Minimize to Reduce Wall Weight)	Per E-D Nozzle Proposal LR711618 and O Chamber Fab & Testing
	e. Gas Side Wall Thickness	.030 in. at Throat	Per E-D Nozzle Proposal LR711618 and O Chamber Fab & Testing
11.	Method of Attachment to Injector	Bolted Flange with Metal Seal.	Desirable for Component Inspection and Refurbishment Coats
12.	Method of Attachment to Nozzle	Braze	Least Wt., Reliable Seal, N.F. Chamber N.F. Nozzle
13.	$P_{chamb.}$ Inlet/P_c	2.0 or 2.4	Pressure Schedule Requirement (Turbine Inj + Coolant ΔPs)
14.	LCF Safety Factor (Design to Unmanned reqmt)	4.0 (Manned) 2.0 (Unmanned)	SSE Data, Historical Data, SF Strain = SFNF = $(1.4)^2 = 2.0$ Additional SF = 2.0 for manned
15.	NF Design (Design to Unmanned Reqmt)	1200 (Manned) 600 (Unmanned)	300 Cycle Requirement + Item 14.
16.	Static Safety Factors	See Table	Historical Determination
17.	Wall Thermal Gradients	$\Delta T_{wall} = 630^oF$ for N.F.S.F = 4.0 $\Delta T_{wall} = 930^oF$ for N.F.S.F. = 2.0	See LCF NF vs ΔT_{wall} Graph

TABLE XV (cont.)

II. NOZZLE

	ITEM	ASSUMPTIONS	JUSTIFICATIONS
1.	Type	Fuel Regen. Brazed Tube Bundle Two Passes	Prelim. Study, Design Flexibility, Low TCA Wt.
2.	Material	ARMCO 22-13-5	Data from SSE Studies - Good LCF vs Hold Time Above 1100°F, Lower P_c Than SSE; Can Repair & Braze
3.	Tube Configuration	All Identical Tapered Tubes, Tapered Wall, Round Tubes, (Unflattened)	Good Heat Transfer Prop., State-of-the-Art, Least Wt. for Max. Strength
4.	Tube Bifurcations	As Required for Cooling	Design Flexibility for Study, No Fab Difficulties
5.	Nozzle Inlet, ϵ_1	To be determined	Limited by LCF life, Material Strength @ Temp., and Heat Flux, e.g., P_c & ΔP & ΔT Coolant
6.	Coolant Inlet + (Torus), ϵ_3	To be determined	Limited by Low Heat Flux Capability of Cold H_2 Inlet and One Tube Design
7.	Nozzle Exit, ϵ_2	@ ½ Engine Length, or ½ Nozzle Length for Larger Thrust Engines (see Graph)	Preliminary Tradeoff Study Results for Max. Payload (Low Weight & Min. Stowed Length)
8.	Chamber Attachment Method	Brazed	Least Wt., Reliable Seal
9.	Method of Attachment	Flange & Seal with Provision for Hot Gas Dump	Must Accept Retractable Nozzle with Cooled Flange
10.	Coolant Turn Around at Aft End	Copper	Low Weight Manifold - Integral with Skirt Flange, Well Cooled @ Tip
11.	LCF Life	300 with 4.0 = 1200 Predicted (Manned) and 300 x 2.0 = 600 (Unmanned) (Design to Unmanned Cond.)	Reqmt. + See Chamber Justification

TABLE XV (cont.)

	ITEM	ASSUMPTIONS	JUSTIFICATIONS
12.	Tube Strain at Crown (Max.)	1.3% @ NF = 1200 & 1.45% @ NF = 600	Based on SSE & Prelim. OOS Calcs, & NF (See #17 & #18)
13.	Gas Side Wall Temperature	1200°F for NF = 1200 & 1400°F for NF = 600	Based on SSE & Prelim. OOS Calcs, & NF (See #17 & #18)
14.	Minimum Wall Thickness	.015 in.	Assumed Fab. Limit
15.	Single Tube Section Wall Thickness Range	2.0/1	Fab Experience
16.	Single Tube Section Diameter Range	3.5/1	Fab. Experience
17.	Coolant Outlet Temp	-150°F	Heat Transfer Analysis
18.	ΔT_{wall}	600°F	Heat Transfer Analysis

TABLE XV (cont.)

III. SKIRT

ITEM	ASSUMPTIONS	JUSTIFICATIONS
1. Type	Extendable/Retractable*	Packaging Requirement & High I_s goal
2. Primary Cooling	Radiation	Simplicity, Low Weight
3. Supplemental Cooling	Maintain Flange Temp @ 1500°F	Maintain Material Structural Properties in Flange to Reduce Weight
4. Material**	AGCarb-101	Apollo Data, SSE Data, Nerva Data, Flexibility of $\mathcal{E}_2$ for Study, Same Wt. as Cb-103
5. Wall Thickness	Variable $\leq$ .20 in.	Min. Wt, Adequate Structural Stiffness and Strength, Min. Fab. Thickness
6. Contour	Rao. Min. Length Contour Is Optimum	Max. Engine Dimensions Limit & Max Performance
7. Entrance Area Ratio, $\mathcal{E}_2$	Same as for Nozzle Exit	Same as for Nozzle Exit
8. Support Brackets	Three Min.	Attachments for Extend/Retract Mechanism
9. Stiffening Rings	Two Min.	One for Attachment of Support Bracket and One at Aft End to Prevent Buckling

* Stowed Position in OOS (Retracted) of Skirt Length is 82 in. from Gimbal Point; 15 in. Length Assumed for Gimbal to Throat Length for parameter analysis.
** Alternate Material: Cb-103 Where Engine Cycle Coolant & P_c and $\mathcal{E}2$ Allows Use. T_w max = 2300°F.

TABLE XV (cont.)

IV. MAIN INJECTOR

ITEM	ASSUMPTIONS	JUSTIFICATIONS
1. State of Propellants	Gaseous Fuel/Liquid Oxidizer	No O_2 Heat Exchanger Required. Good Idle Mode MR Control. Good for Staged Combustion Cycle, Short L'.
2. Type	Platelet Vane	Low Blockage Ratio Best for Gas/Liquid With Staged Combustion; Ref SSE, Ares & See #4.
3. Material	Body ARMCO 22-13-5, Vanes CRES 347, Ni 200	High Strength, Brazable, Thermal Cond. Control
4. Injection Pattern	Impinging Oxid. Doublets, Shower Head Fuel	Short L' Because of Impinging Elements & Small F/E.
5. $\Delta P/P_c$ @ Full Thrust	Oxid = .65, Fuel = .06 (Min. Ox. ΔP = 400 psia)	Min. for 5/1 Throttling Stability* SSE Data
6. Manifolding	Oxid Torus (ΔP man = .05 P_c or 30 psia, Whichever is Greater)	Distribute Oxid. Around O.D. to Vanes, Weight ΔP Tradeoff
7. Blockage Ratio (Vane Area/Open Area)	.42 for S.C. Cycles, Larger for Other Cycles depending on Injector Concept	SSE Ratio, Gives Desired Fuel Injection Velocities
8. Attachment to Chamber	Bolted Flange	Desirable for Component Inspection and Refurbishment. Very Small Loads for high ϵ_o Nozzle
9. Attachment to Hot Gas Manifold	Bolted Flange	Same as Above Except Large Loads. Large, Bulky, Awkward HGM to Handle if Injector Were Welded to and Machined from HGM During Refurbishment
10. Gas Distribution Device	Separately Located Upstream at HGM Interface	Required for Optimum Gas Distribution and Injector ΔP

* For 10:1 Throttling, Staged Combustion Cycle Use Same Oxid. System and Let ΔP Go Up. (Check Strength of Vanes at Increased ΔP.)

TABLE XV (cont.)

V. PRE-BURNER

	ITEM	ASSUMPTIONS	JUSTIFICATIONS
1.	Injector Type	Platelet, Dual Oxid. Circuit	High Perf., Throttlability, Short Length
2.	Injector Material	Ni-200 Platelets	Good Thermal Conduction at Face
3.	Chamber Material	Inconel 718	High Strength @ Elevated Temp
4.	L'_{gg} (in.) Length	6 in.	SSE Data
5.	T_{ti} Turbine Inlet Temp.	1860°R	Material Limit for Cycle Life Reg.
6.	$C.R._{gg}$ Contraction Retro.	3.5/1 (to Turbine Nozzle)	ΔP_{ch} = 2.0% of P_{cgg} (Chamber Pressure Drawings)

TABLE XV (cont.)

VI. IGNITERS

MAIN INJECTOR & PREBURNER/GAS GENERATOR

ITEM	ASSUMPTIONS	JUSTIFICATIONS
1. Type	Electric Torch	Reliable, Demonstrated
2. Initiator	Spark	Reliable, Demonstrated
3. Materials	Electrode N_i 200; Chamber N_i 200	APS Ignition Program
4. Seal	Electrode Housing Brazed Metal to Ceramic	Has Been Demonstrated (J-2 Plugs)
5. Spark Energy	5 Modules/Spark	APS Data
6. Spark Rate	100 Sparks/sec $\approx$ 10 Mil Ign. Delay	Test Data Available @ 50 & 500 Sparks/Sec
7. Spark Voltage	20KV	APS Experience
8. W_t Torch Flow	$.065\ W_o$; $.0125\ W_f$	APS Experience
9. Min. Temp of Torch	$1100^{\circ}F$	Auto Ignit. Temp of H_2 O_2
10. Electrode	O_2 Cooled, Submerged	APS + Main Engine Ignit. Experience

TABLE XV (cont.)

VII. TURBOPUMPS

	ITEM	ASSUMPTIONS	JUSTIFICATION
1.	Type	Each pump driven by its own turbine	Turbopump envelope and space availability speed incompatibility.
2.	Low Speed Pumps	Hydraulic Drive	Specified NPSH (60 and 16 feet for LH_2 and LO_2 respectively).
3.	Bearing	Rolling element, propellant lubricated bearings. DN equal approximately 1.5 x 10^6 and 1.0 for LH_2 and LO_2 respectively.	Demonstrated 23 hours with LH_2 ball bearings at 1.8 x 10^6 DN. Demonstrated 15 hours with LO_2 ball bearings at 1.12 x 10^6 DN. (Above times were without failure.)
4.	Pump Suction Specific Speed	Design S = 45000 corresponding to induce cavitating to non-cavitating head ratio equals 20%.	Current Design Point for Rocket Engines. Technology demonstrated at S = 55000 with water.
5.	Pump Suction Safety Factor	Safety factor = 1.1.	Pump design shaft speeds are reduced by dividing the available effective NPSH by the safety factor. (Effective NPSH equals NPSH + Thermodynamic suppression head.)
6.	Pump Stages	One stage, LO_2 pump. Two stage, LH_2 pump.	Current Rocket Engine design approach.
7.	Pump Recirculation Flows	Allocated 5% in each pump stage.	
8.	Pump Efficiency	Computed values from curve fit.	Pump Efficiency Curve fit depicts rocket engine pump efficiency values as function of specific speed and flow values.
9.	Pump Material	Aluminum impeller and housings.	Light Weight, high strength to weight ratio.

ITEM	ASSUMPTIONS	JUSTIFICATION
. Turbine Temperature	1860°R (1400°F)	Typical turbine material (waspaloy) retains strength properties to this value. Turbine temperature increases to 2000°R will increase delivered specific power approximately 3% while reducing low cycle fatigue life by a factor of 10.
11. Turbine Rotor nozzle cooling.	None	Turbine gas temperature selected sufficiently low to eliminate requirement.
12. Turbine Stages	see below	Acceptable Rocket Engine Design approach.
13. Turbine Pressure Ratio	Computed for topping cycles as required to satisfy power balance except for stage combustion bleed where oxidizer turbine pressure was set equal to 20:1. Bleed cycle pressure ratios were 4:1 and 15:1, LO_2 and LH_2 turbines respectively.	Oxidizer Turbine Flow are typically low and result in partial admission which in turn suggests the use of a single stage turbine. A pressure ratio of 4 was selected. This value does not affect turbine flow in the series turbine bleed cycle systems.
14. Turbine Mean Blade Speed	Fuel Turbine, 1500 ft/sec. Oxidizer Turbine, 1000 ft/sec.	Acceptable Rocket Engine Mean Blade speed on hydrogen turbine. Oxidizer turbine mean blade speed reduced to reduce turbine rotor diameter and polar moment of inertia to equalize pump spin up times.
15. Turbine Nozzle Angle	15°	Minimum practicle angle from mechanical design considerations. Small angle selected to achieve maximum efficiency.

Turbine Stages (item 12):

	Oxid Turb.	Fuel Turb.
Expander - Parallel Turbine	1	2
Stage Combustion Parallel Turbine	1	2
Stage Combustion Bleed	1	2
Gas Gen. - Series Turbine	2	2
Combustion Gas Topoff - Series Turb	2	2
Coolant Tapoff - Series Turbine	2	2

TABLE XV (cont.)

[illegible]. TURBOPUMPS (Cont.)

ITEM		ASSUMPTIONS	JUSTIFICATION
[illegible].	Turbine Hub/Tip Ratio	0.8 Maximum	Turbine Rotor Hub to Tip Diameter Ratio selected at 0.8 maximum to limit tip leakage losses. Turbines with low flow will be partial admission to maintain this hub ratio.
[illegible].	Turbine Rotor Material	Waspaloy	Material selected for its high ductility characteristics coupled with adequate high temperature strength properties.
18.	Turbine Efficiency	Computed	Computed from a model of turbine design point efficiencies. Efficiency predictions include the effects of turbine nozzle angle and tip leakage.

TABLE XV (cont.)

VIII. VALVES

ITEM	ASSUMPTIONS	JUSTIFICATION
1. Type	Poppet	Positive shutoff selaing. Less opening stroke required. Applicable for line sizes to 2 inches. Above 2 inches consideration would be coaxial poppet, ball, or butterfly.
2. Material		
Body	Stainless Series	Maintain close tolerances at cryogenic temperatures. Machinability and surface good finish control.
Seals (Dynamic and Shutoff)	Plastic or Metal to Metal	Application (temperature, life cycle, pressure, leakage criteria, and seal function) will determine material selection.
3. Actuation	Electric	Fast response is not required. Difficult to obtain positive sealing, at cryogenic temperature, with gas actuation. Electric motor adaptable to modulating valve application.
4. Response	Fast and slow response considered.	Response varies as a function of valve concept, actuation configuration, line pressure, available power to actuator, and valve size.

6. Engine Cycle Evaluation for 8K to 50K Thrust

This section presents the results of the parametric evaluation study of the six engine cycles without consideration of engine design constraints. The purpose of these data is to provide information beyond current engineering design limits and to evaluate payoffs in payload capability through extension of these design limits.

This section is introduced by the result of the JANNAF thrust chamber performance analysis results and forms the basis of the engine performance presentation.

a. Nozzle Contour and Thrust Chamber Performance

The minimum Rao Nozzle contour data used for the parametric performance and weight calculation and the basic parametric thrust chamber performance is presented in Appendix A of this report.

b. Engine Cycle Characterization

The data presented is the result of a parametric study of the six engine cycles. The data is organized by engine cycles and includes:

Delivered vacuum specific impulse.

Engine weight for three different nozzle concepts.

Engine length and maximum engine diameter.

Suction line diameter.

The presented data are not restricted by envelope considerations or engine life requirements but is intended to give general relationships.

The suction line diameter is assumed to be independent of engine cycle and therefore is presented separately.

(1) General Parametric Data

(a) Suction Line Diameter

Suction line internal diameters are shown in Figure 12 in the thrust range from 8K to 50K lb as functions of NPSH at pump suction for engine mixture ratio = 6.

(b) Engine Length and Diameter

For simplicity, the engine length and maximum engine diameter are presented in one figure (Figure 13) which is valid for all engine cycles except the Expander Cycle with a constant length between the gimbal bearing and the throat plane. The data are presented as a function of the thrust to chamber pressure ratio, and area ratio at a mixture ratio = 6.

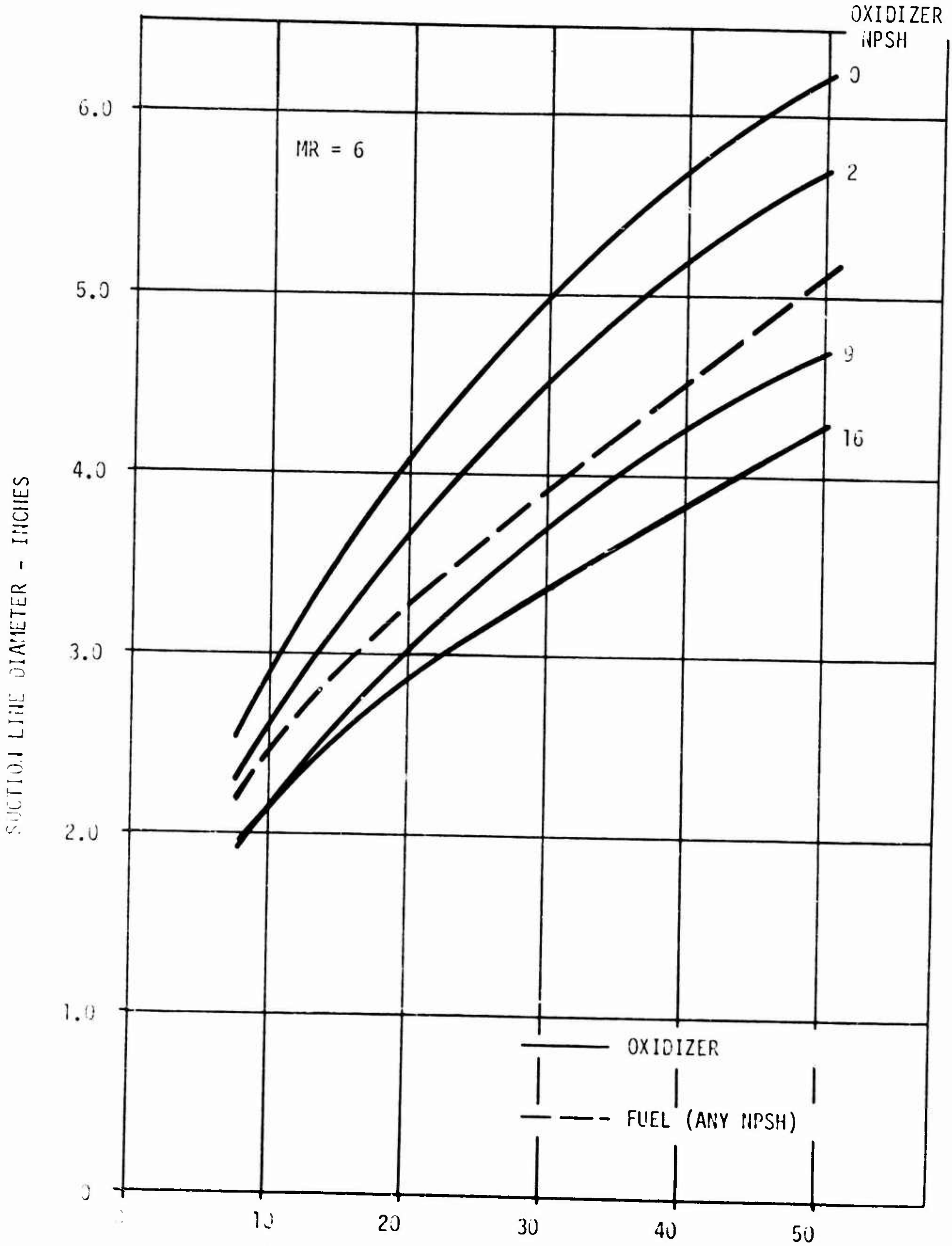


Figure 12. Suction Line Diameter

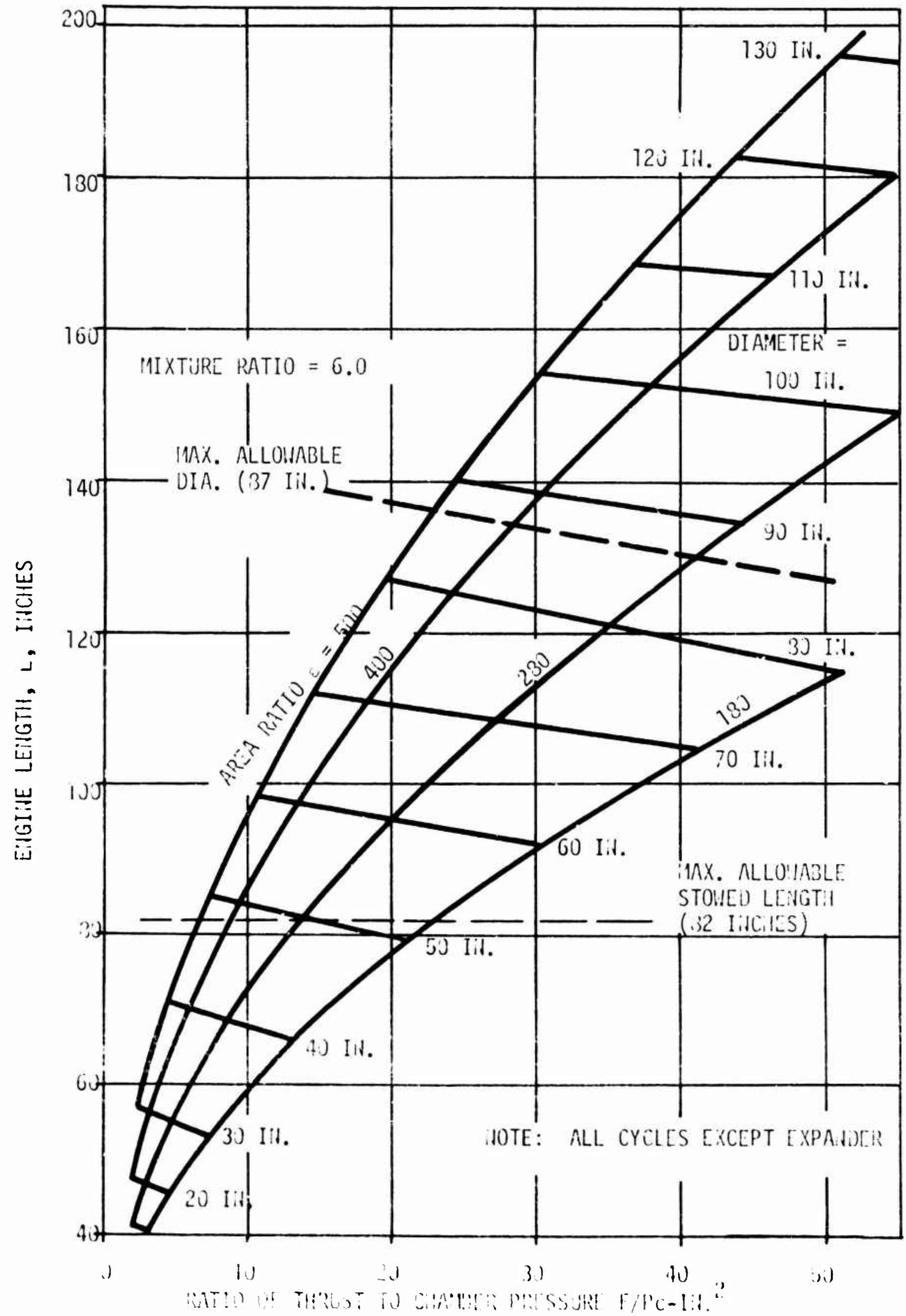


Figure 13. Engine Length, All Cycles Except Expander

The Expander Cycle chamber length is shown separately in Figure 14 for mixture ratio = 6.0. The difference in engine length compared to the other cycles is due to an additional chamber length of 8 in.

Data are also included in this section, for the suction line location relative to the thrust chamber axis as shown in Figure 15.

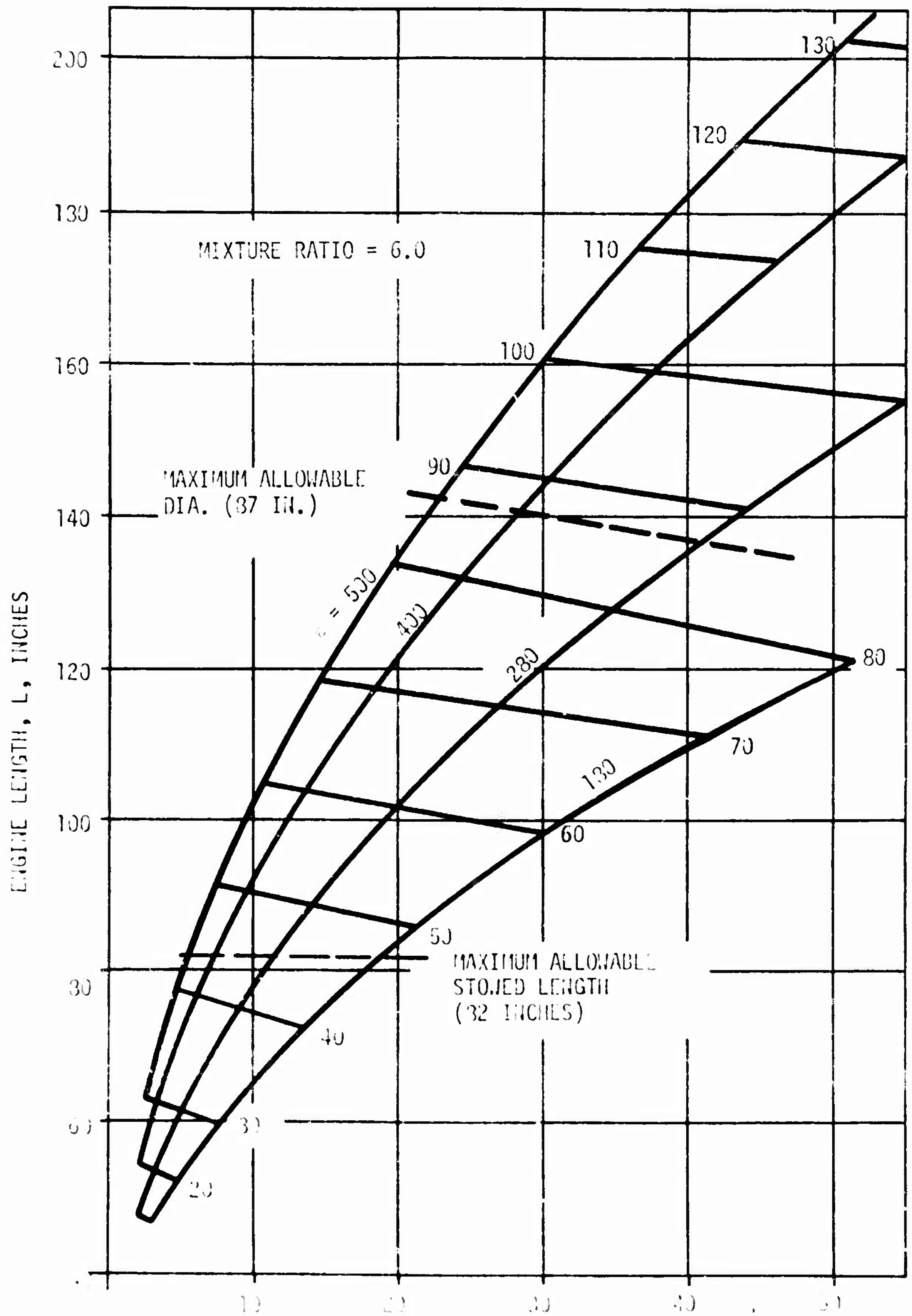


Figure 14. Engine Length, Expander Cycle

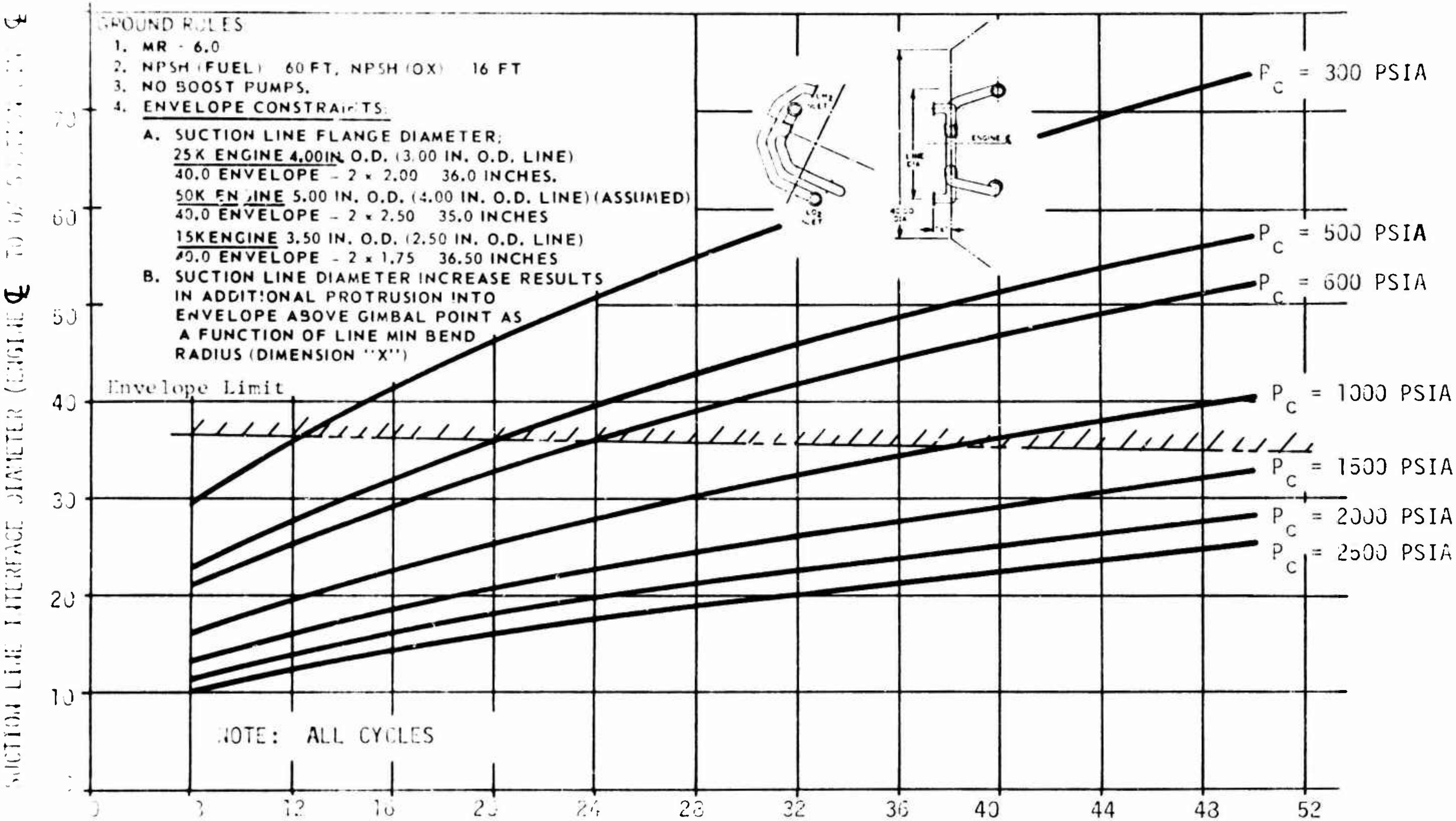


Figure 15. Suction Line Interface Diameter

(2) Gas Generator and Combustion Gas Tap-Off Cycle Characteristics

A configuration drawing of the gas generator bleed cycle engine is shown in Figure 16 and a schematic design is shown in Figure 17.

The performance of the gas generator bleed cycle and combustion chamber tap-off cycle are identical and is shown in Figures 18 through 21. Separate weight charts are presented for each of the two cycles. Figures 22 through 24 show the weights for the gas generator bleed cycle and Figures 25 and 26 summarize the interaction of engine weight and performance for the gas generator bleed cycle.

Figure 27 is a configuration drawing of the chamber gas tap-off engine cycle and Figure 22 is a schematic diagram of that cycle. Figures 29 and 30 are weight data. All weights include low speed pumps.

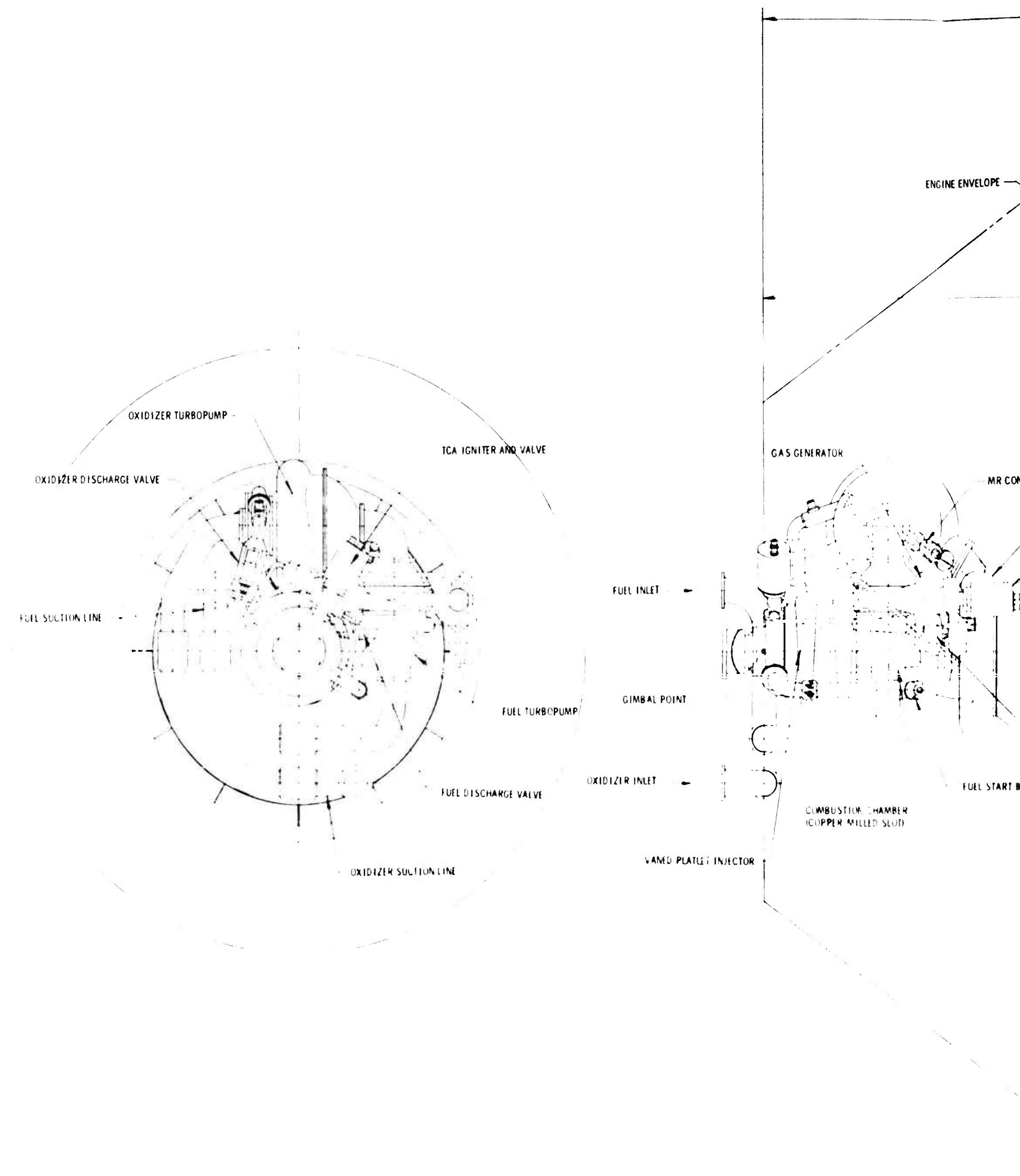
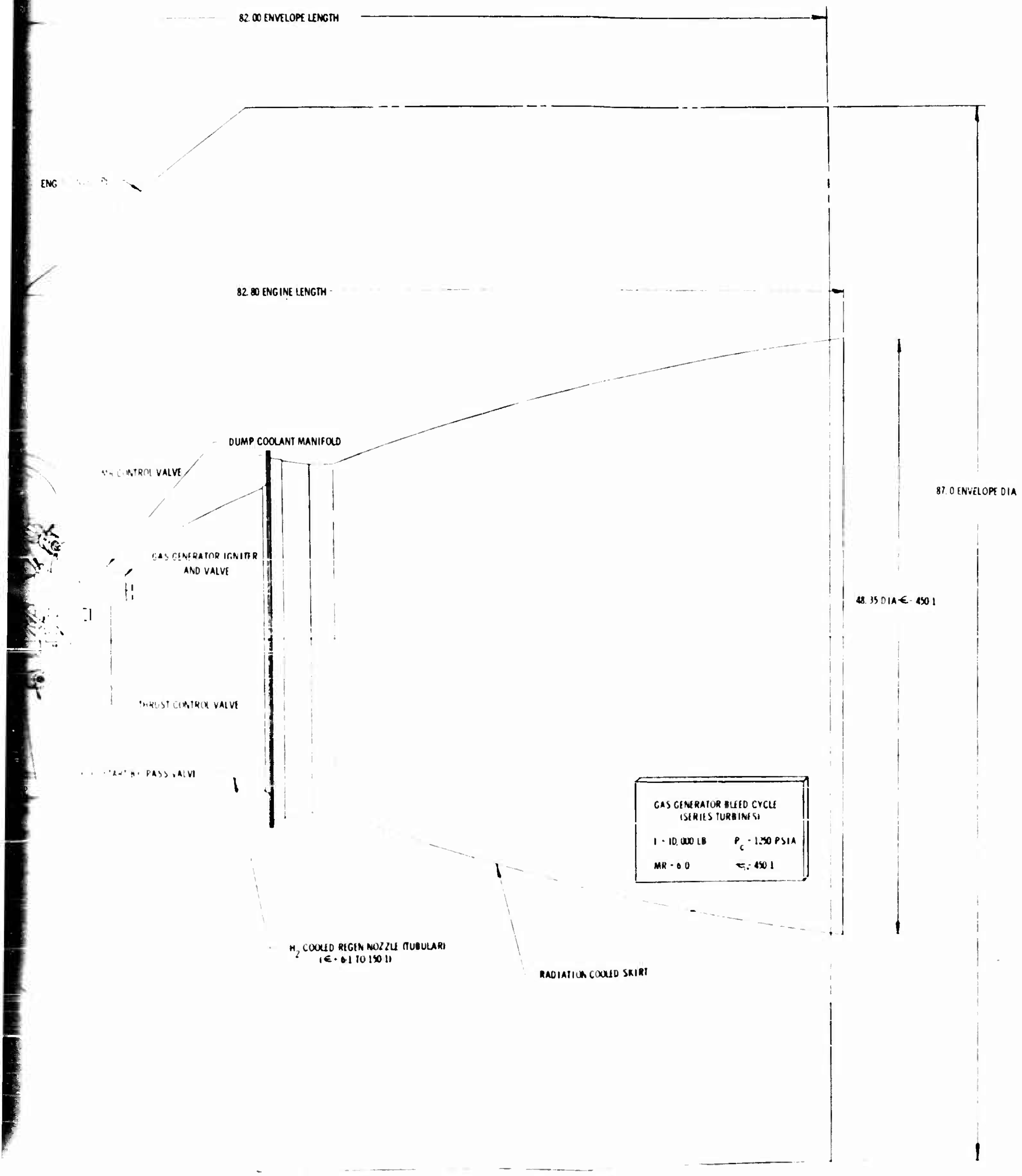


Figure 16. Configur



... Drawing, Gas Generator Bleed Cycle

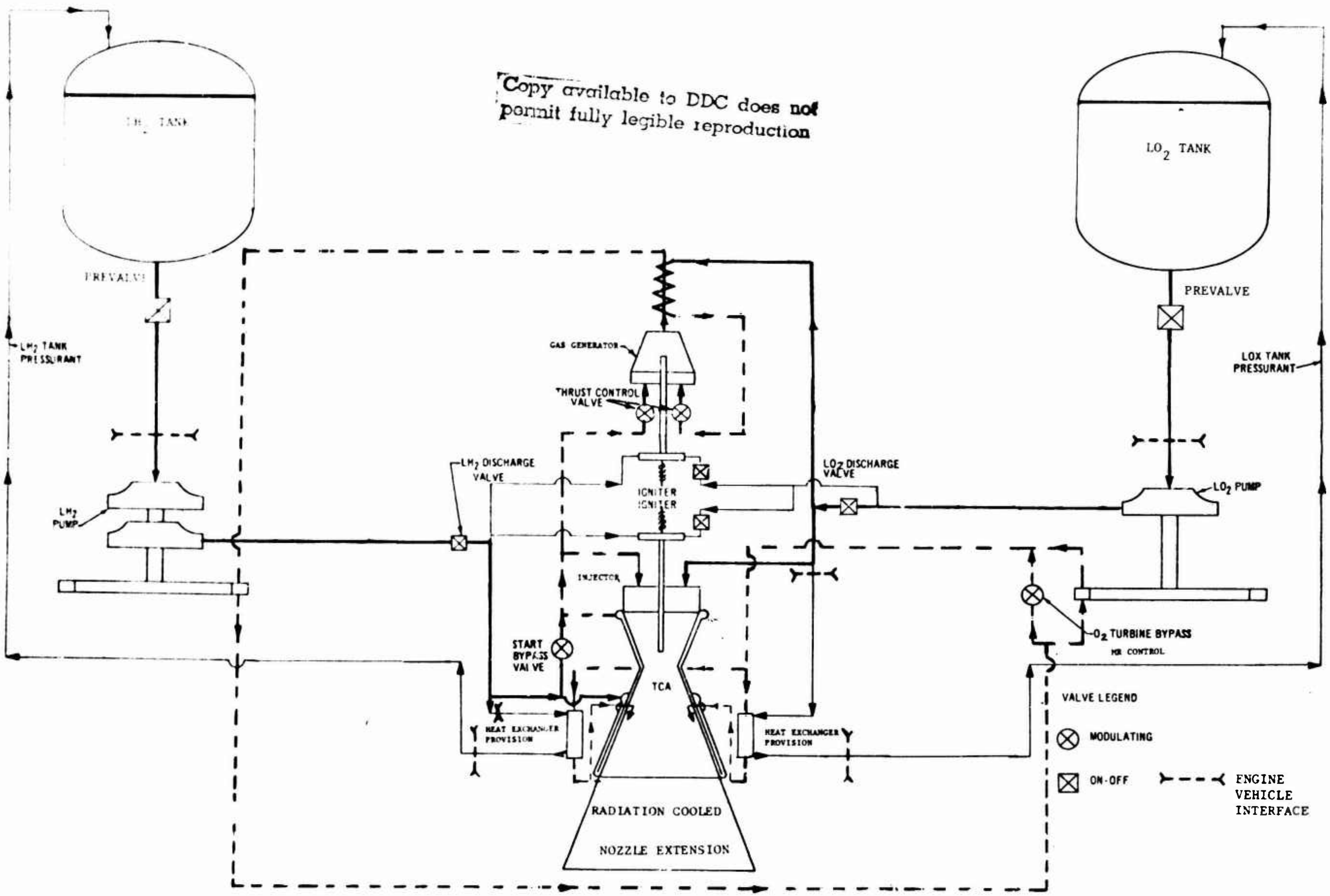


Figure 17. Engine Schematic, Gas Generator Bleed Cycle

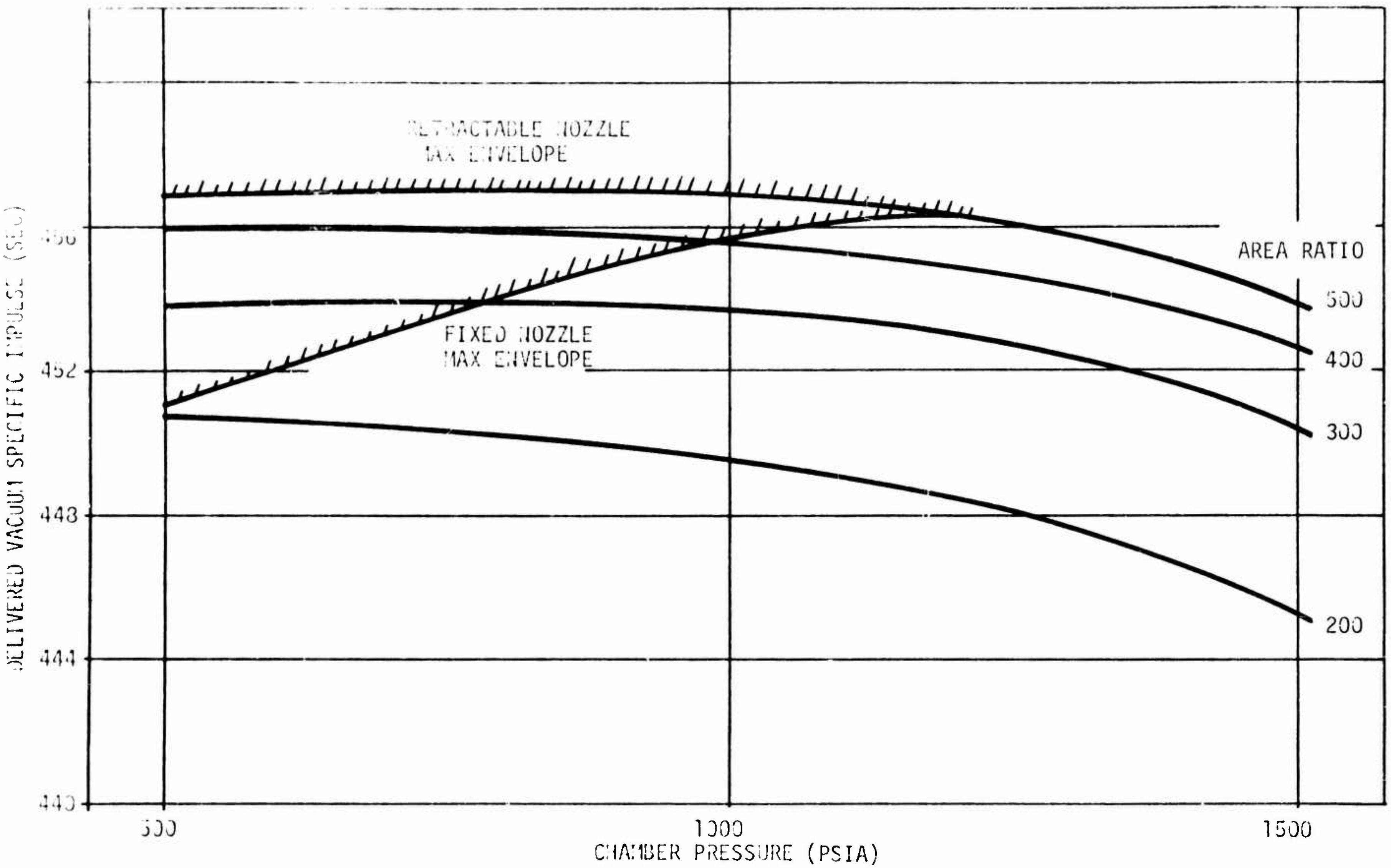


Figure 18. I_s vs P_c, MR = 6, F-8K, Gas Generator Bleed Cycle

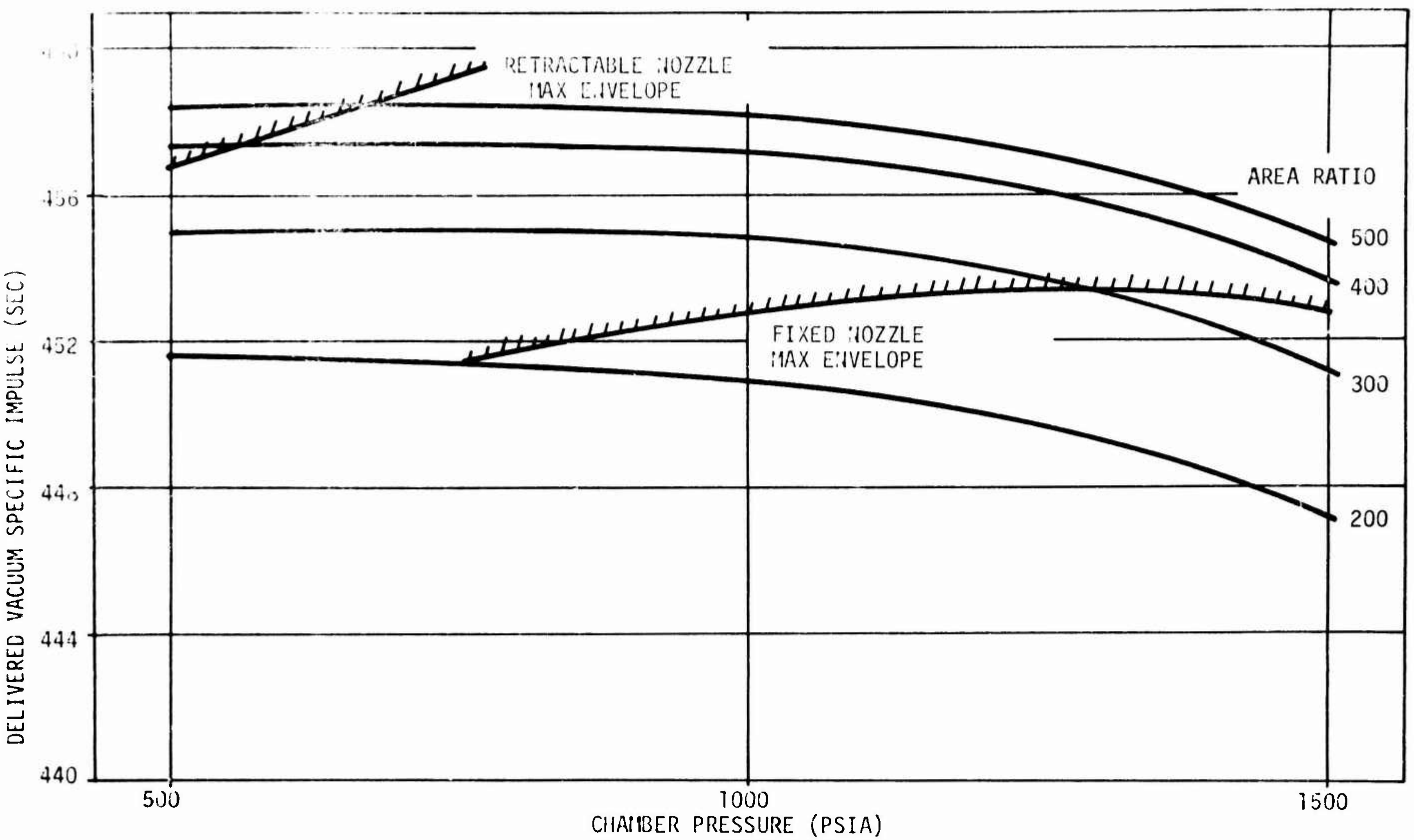


Figure 13. I_s vs P_c, MR = 6, F = 15K, Gas Generator Bleed Cycle

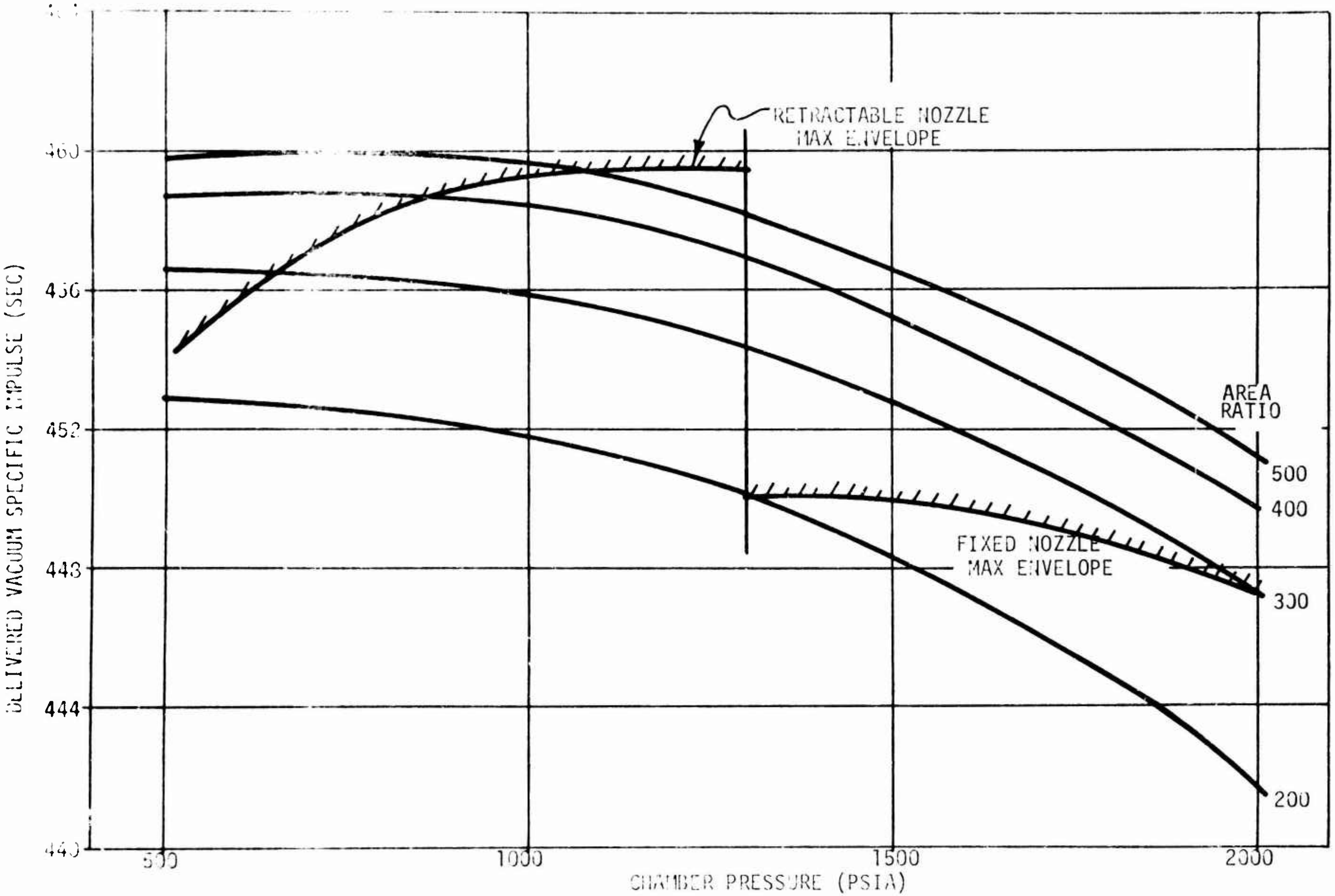


Figure 20. I_s vs P_c, MR = 6, F = 25K, Gas Generator Bleed Cycle

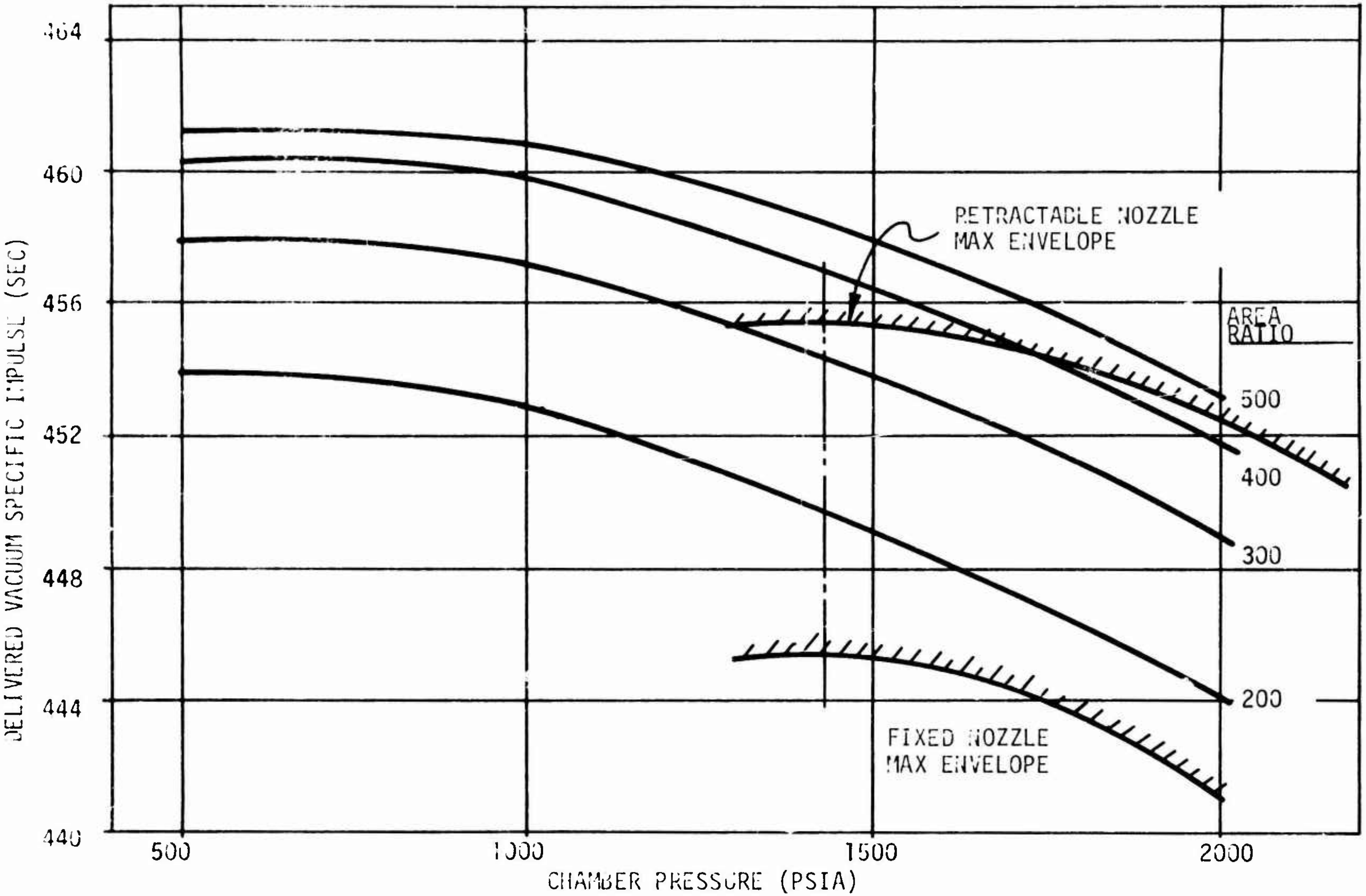


Figure 21. I_s vs P_c, MR = 6, F = 50K, Gas Generator Bleed Cycle

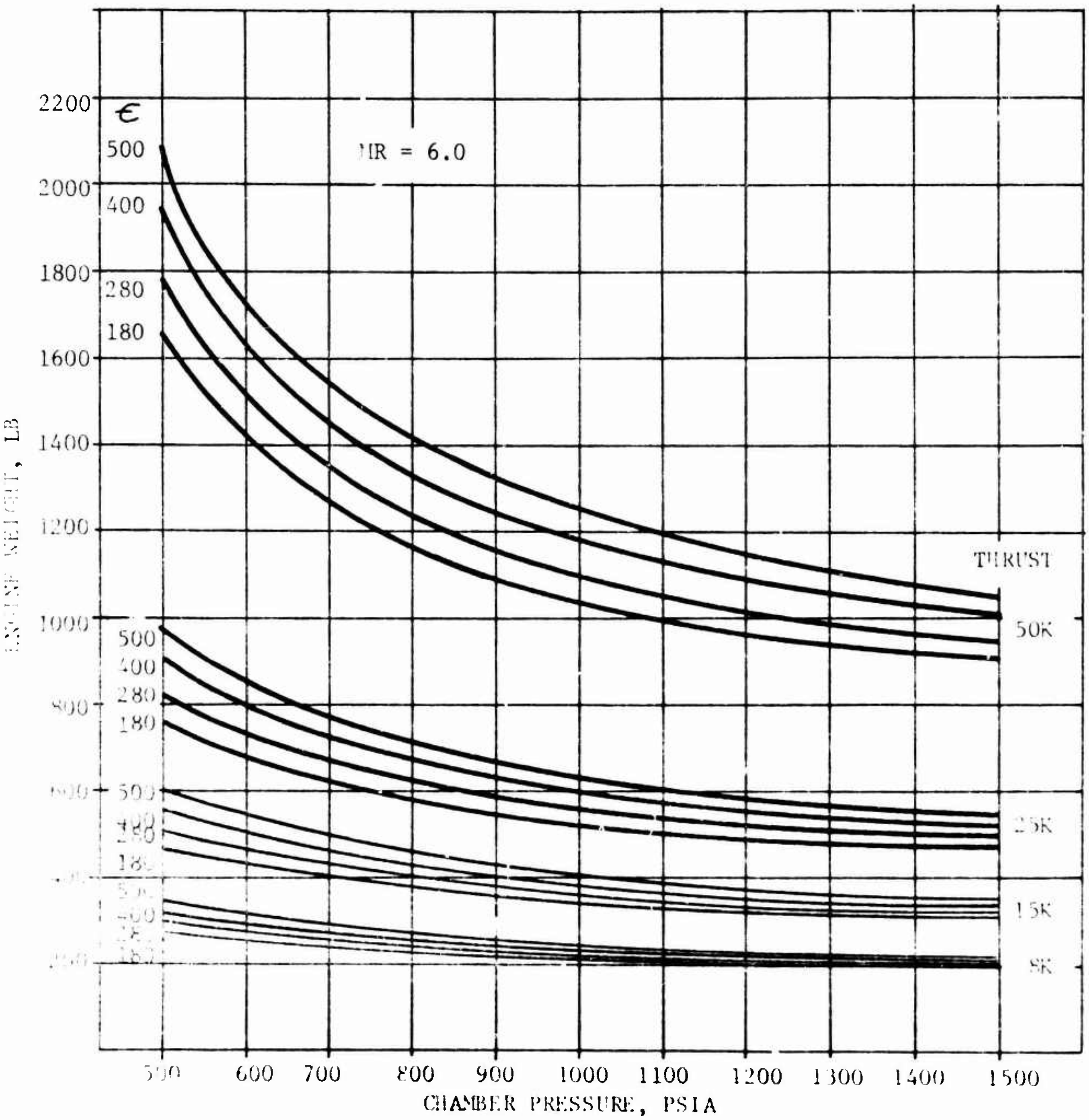


Figure 22. Engine Weight vs P_c, Fixed Nozzle, Gas Generator Bleed Cycle

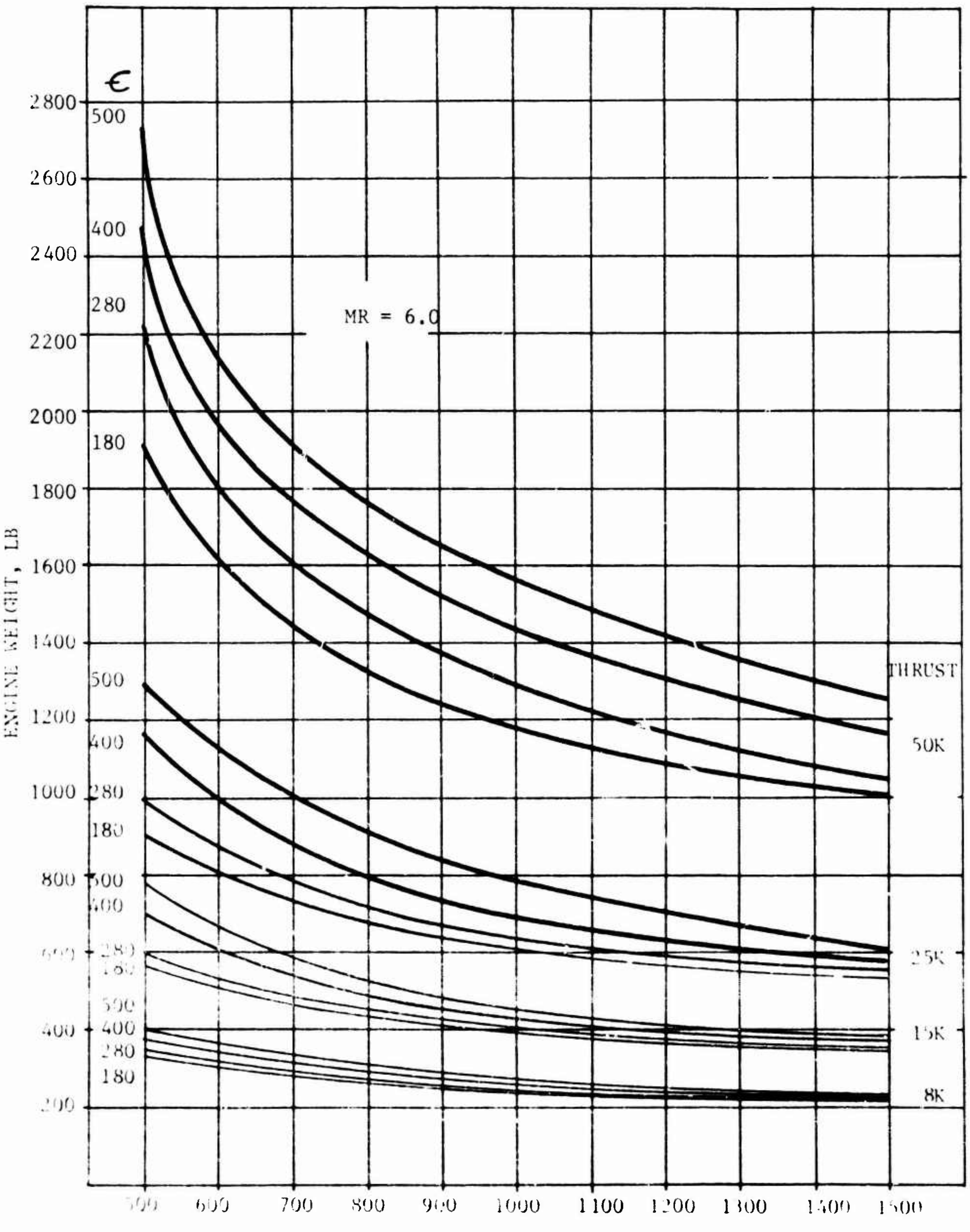
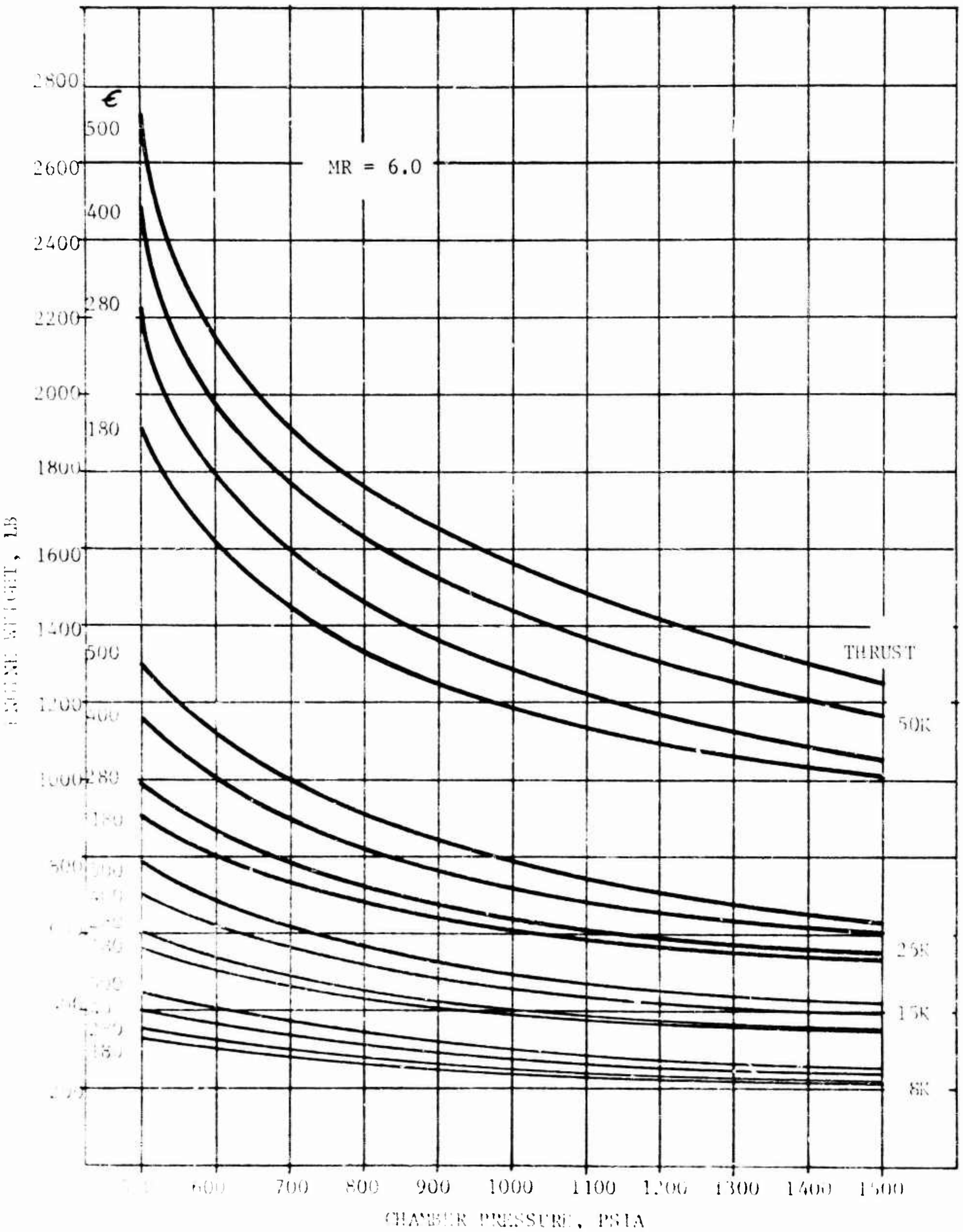


Figure 2?. Engine Weight vs P_c, Minimum Weight Retractable Nozzle, Gas Generator [illegible] Cycle



Engine Weight vs P_c, Minimum Length Retractable Nozzle, Gas Generator Bleed Cycle

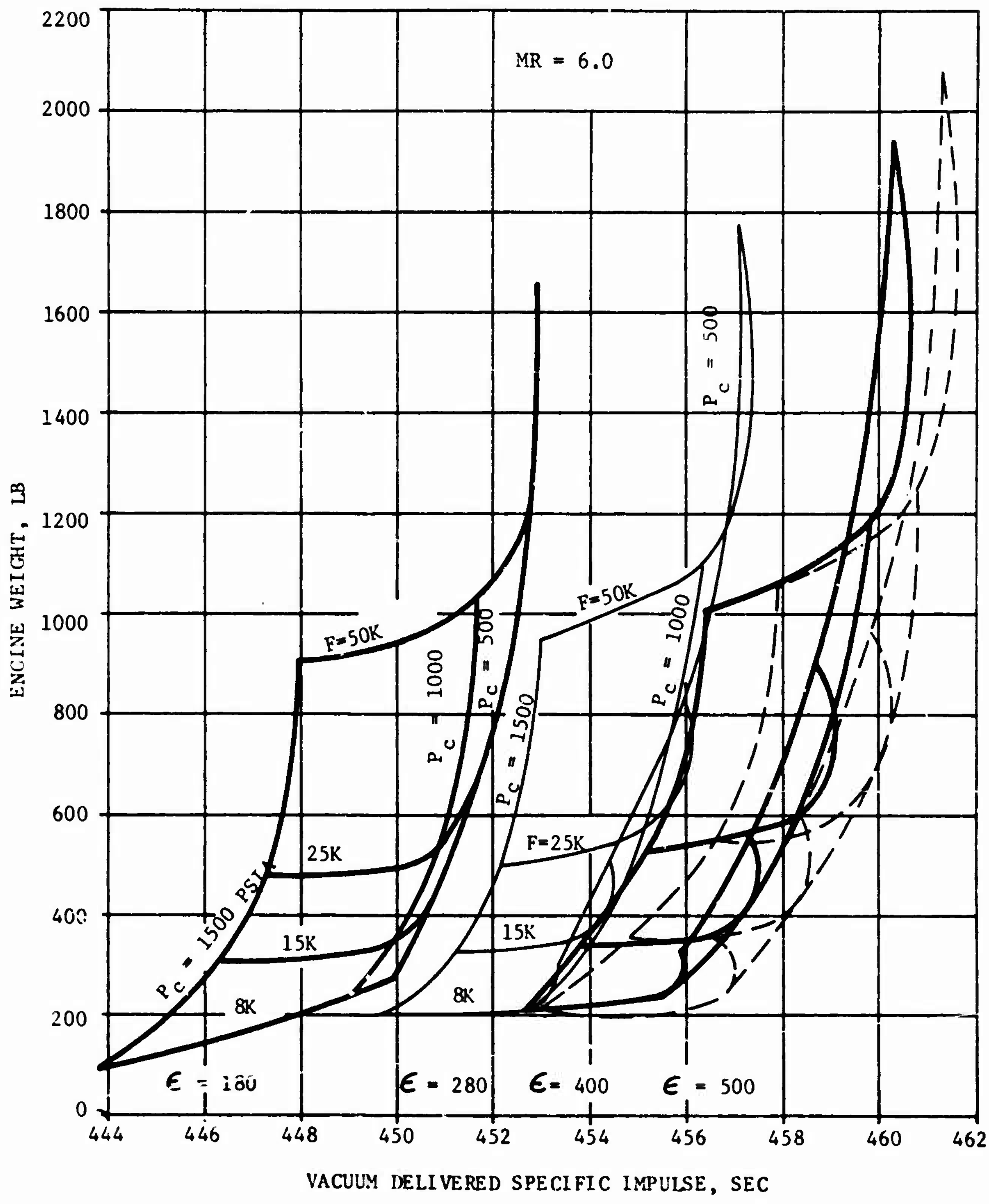


Figure 25. Engine Weight vs I_s, MR = 6 Fixed Nozzle, Gas Generator Bleed Cycle

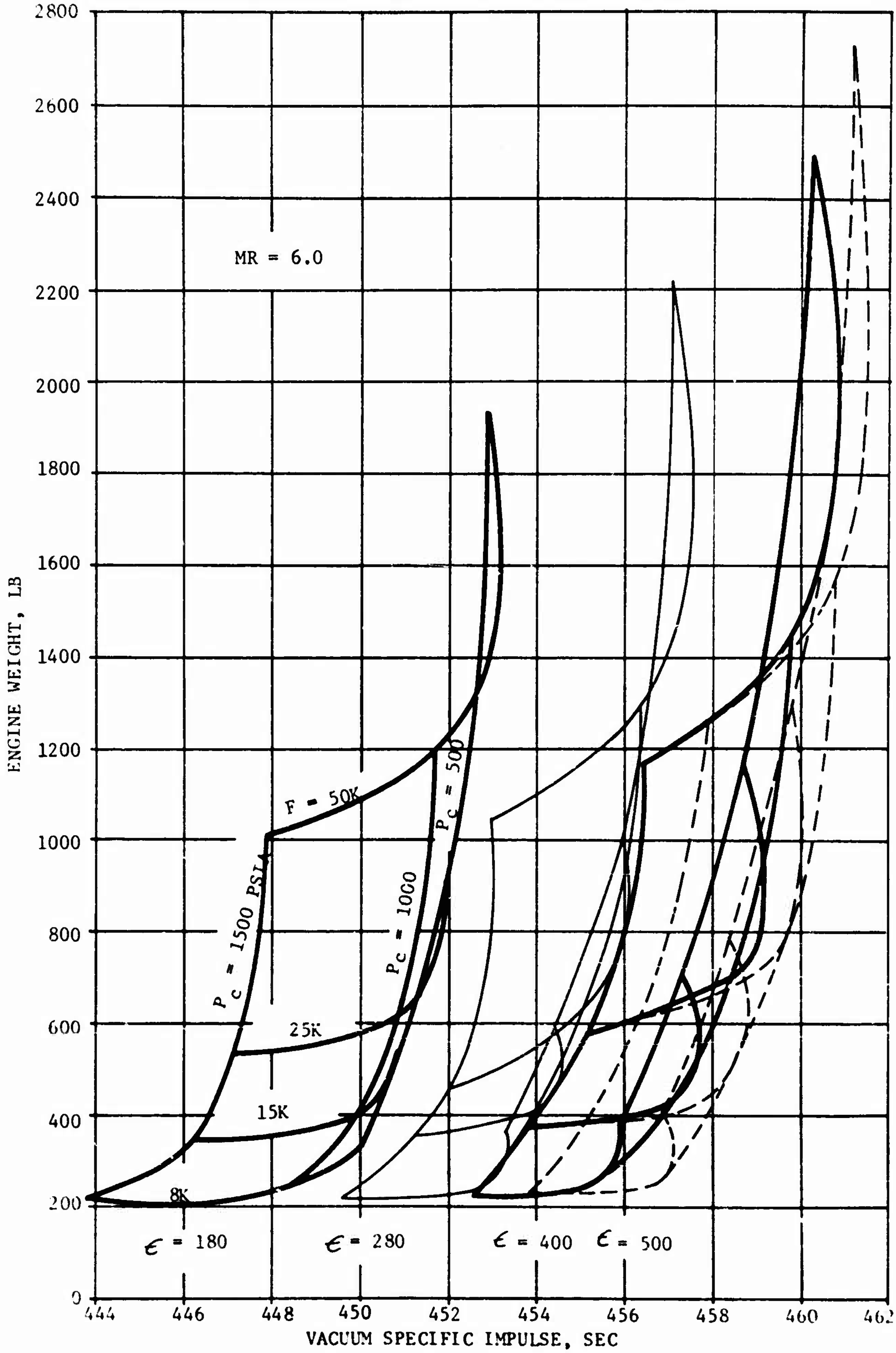


Figure 26. Engine Weight vs I_s, MR = 6, Minimum Weight Retractable Nozzle, Gas Generator Bleed Cycle

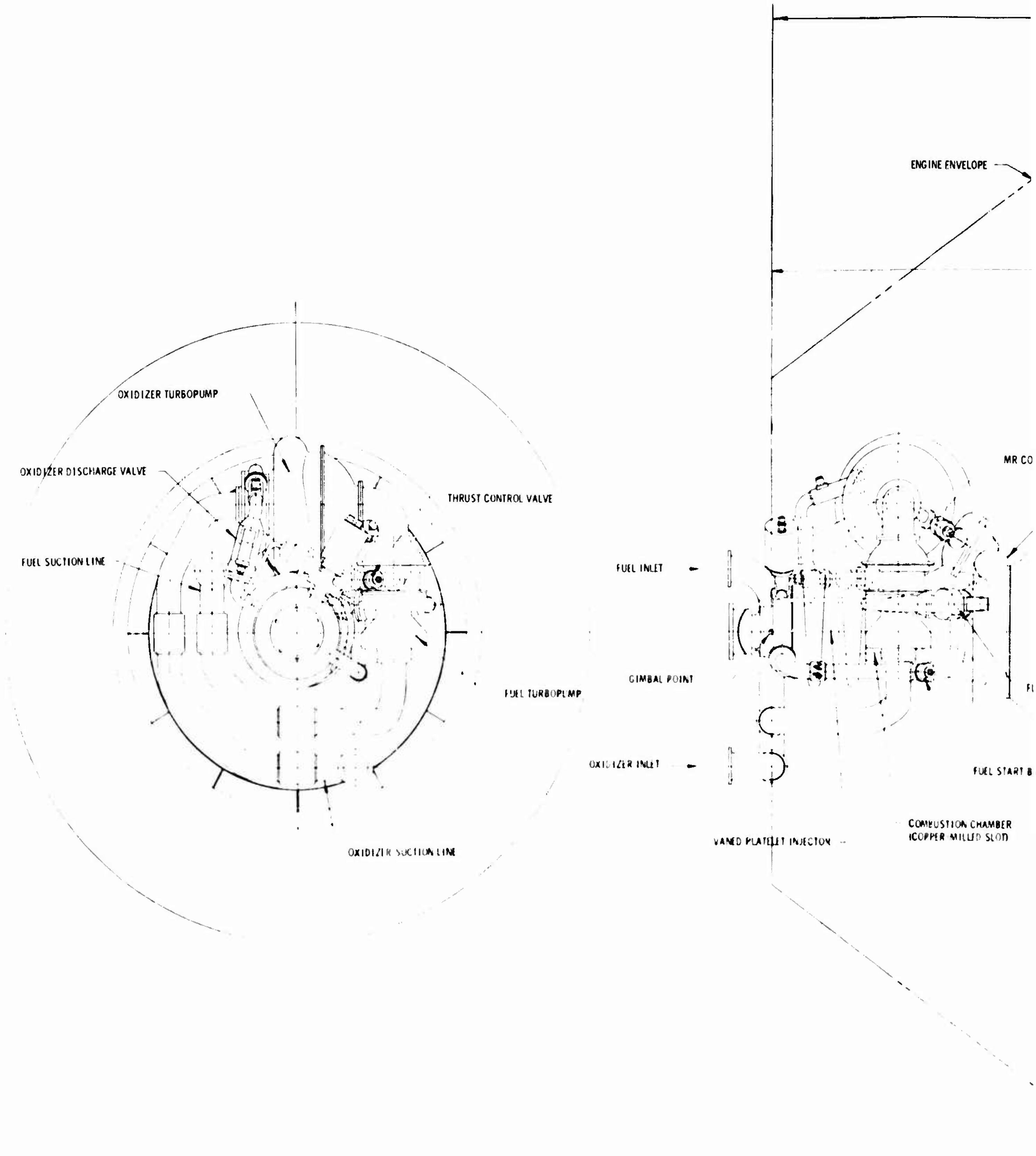
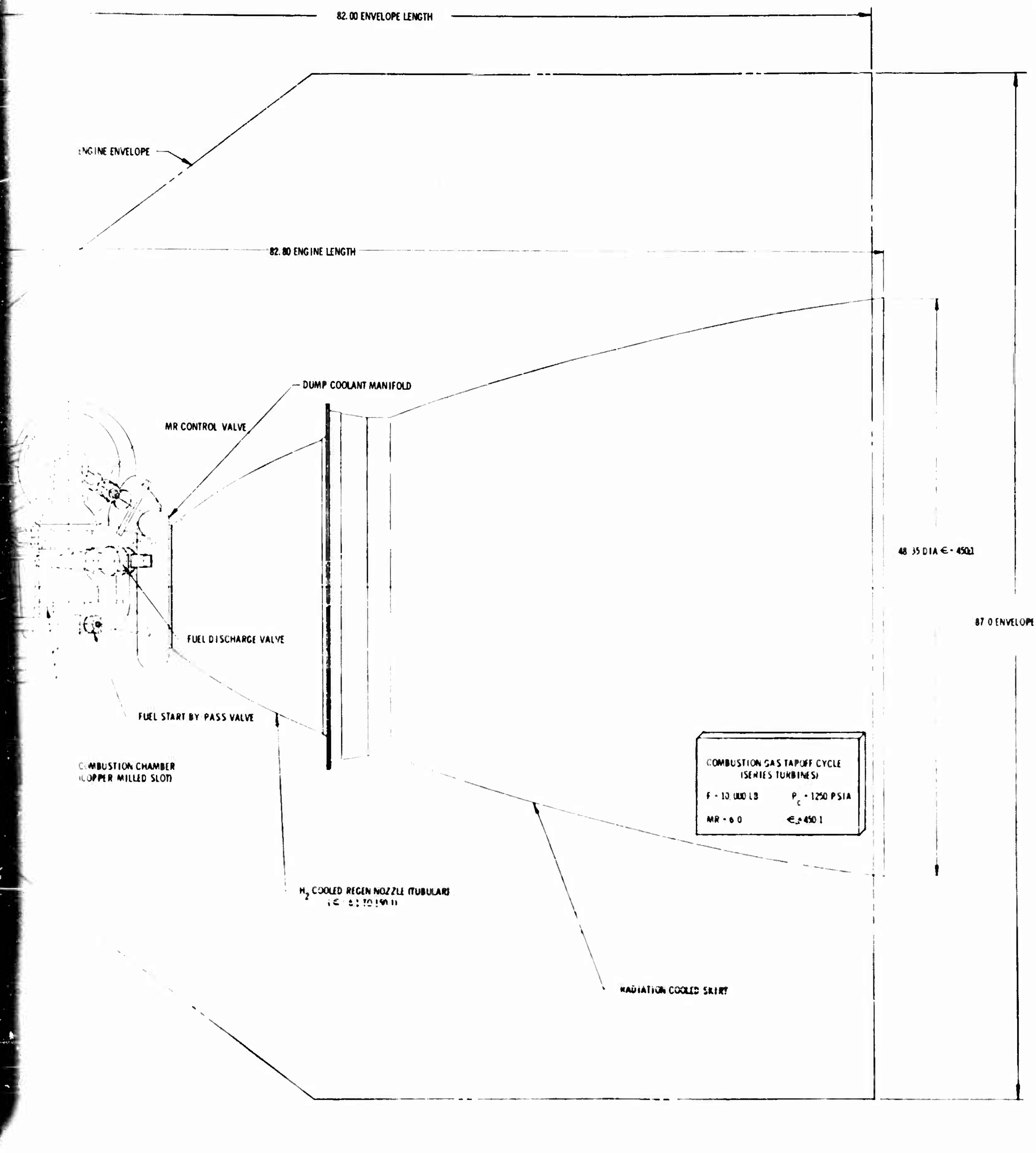


Figure 27. Configuration Drawi



...ration Drawing, Combustion Gas Tapoff Cycle

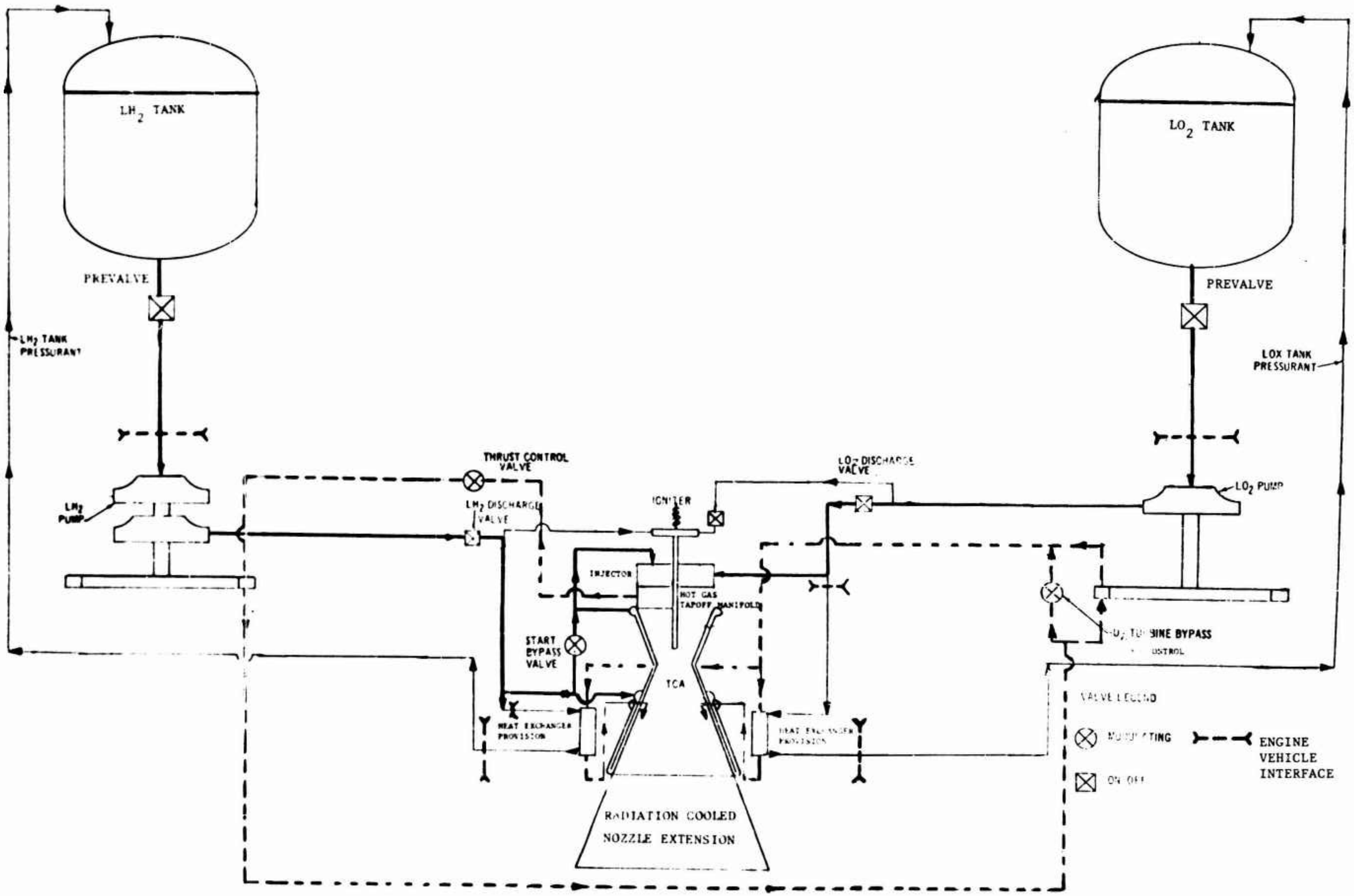


Figure 28. Engine Schematic, Combustion Gas Tapoff Cycle

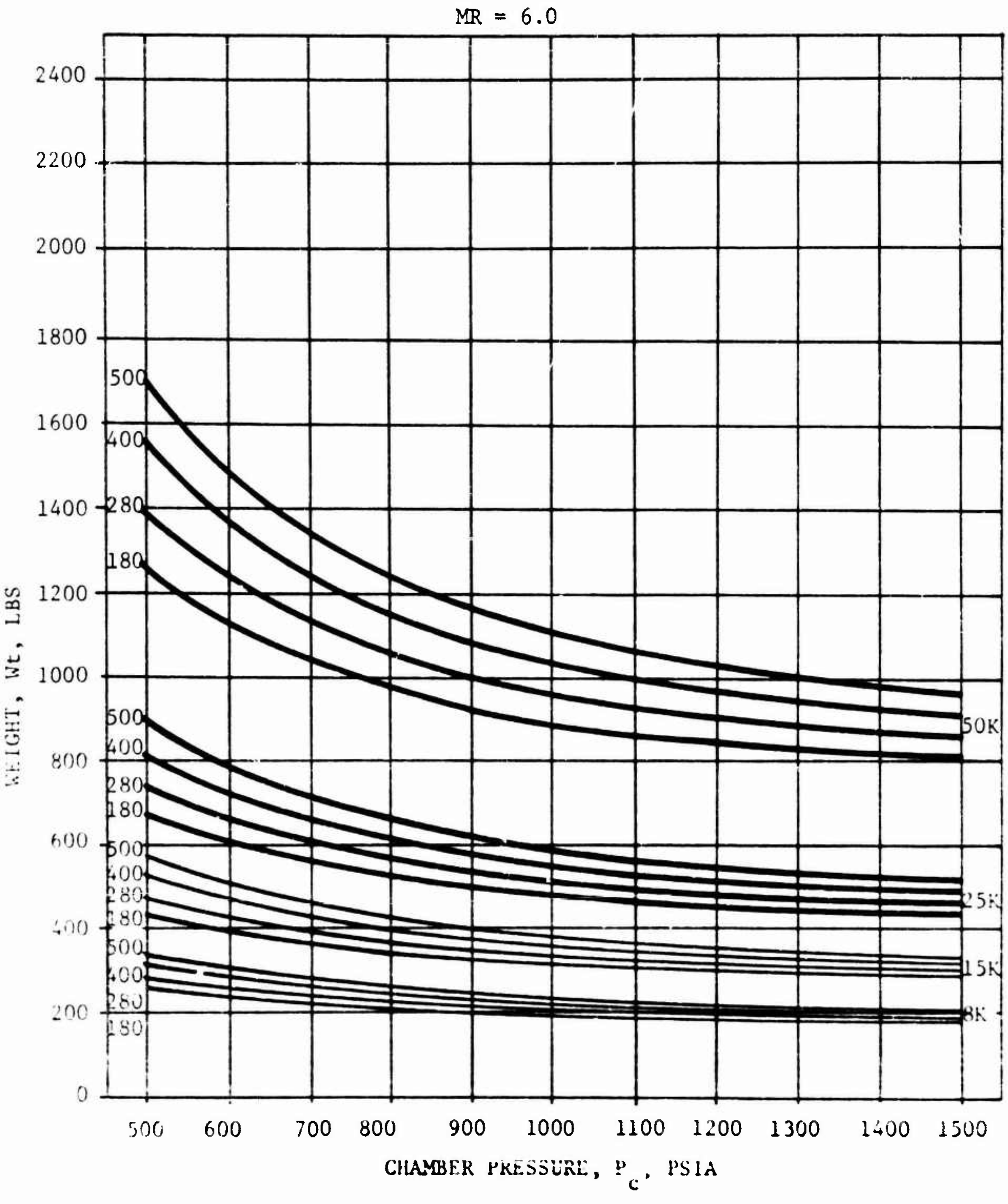


Figure 29. Engine Weight vs P_c, Fixed Nozzle, Chamber Tapoff Cycle

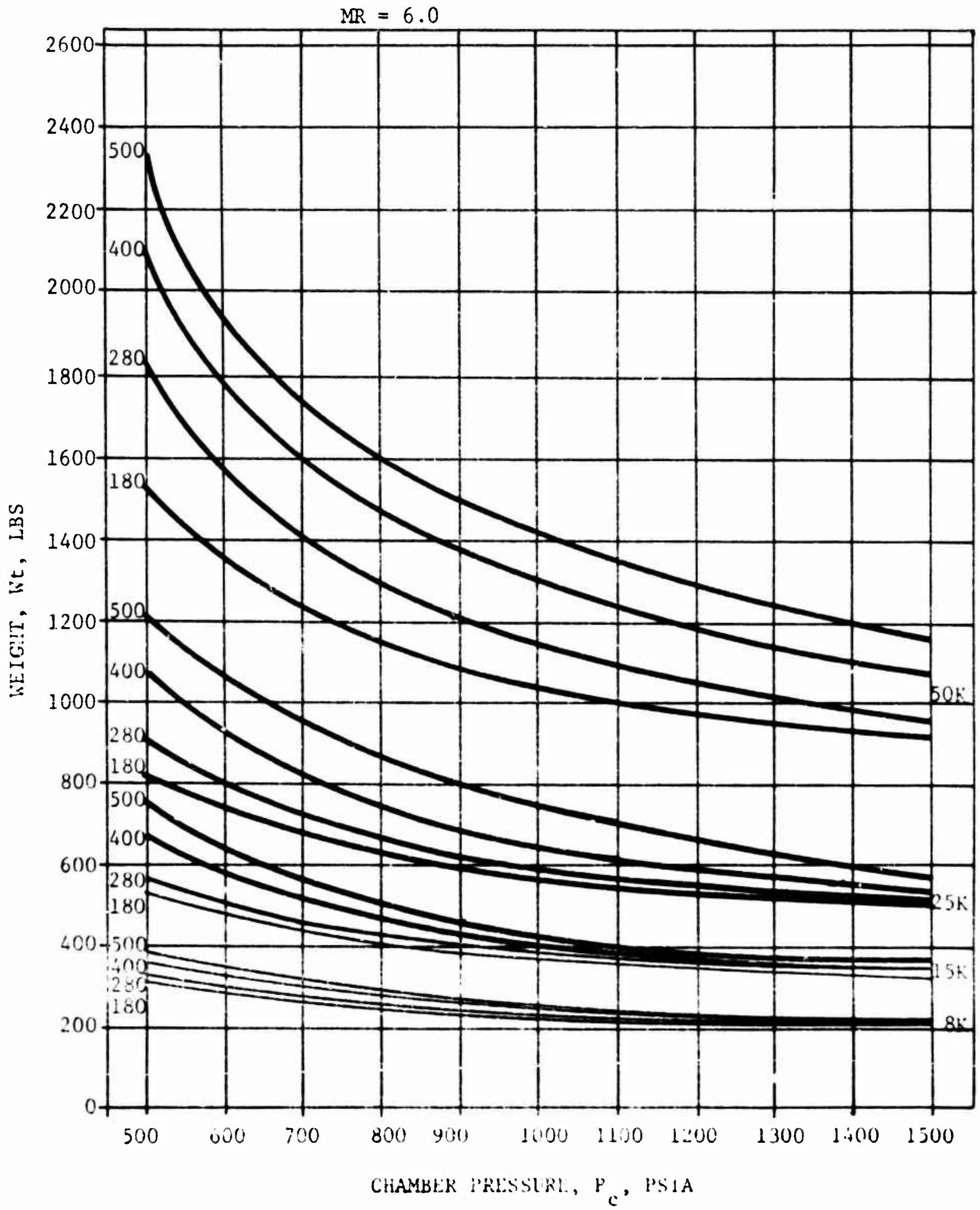


Figure 30. Engine Weight vs P_c, Minimum Weight Retractable Nozzle, Chamber Tapoff Cycle

(3) Coolant Bleed Cycle

This section includes the engine delivered vacuum performances and calculated engine weight as functions of thrust chamber pressure and expansion area ratio including the interaction of weight and engine performance. All weights include low speed pumps.

Figure 31 is a configuration drawing of the coolant bleed cycle engine and Figure 32 is a schematic diagram of the cycle. Figures 33 through 36 show the engine performance. Figures 37 through 39 are weight data for the three nozzle configurations. Figures 40 and 41 depict the interaction of engine weight and performance.

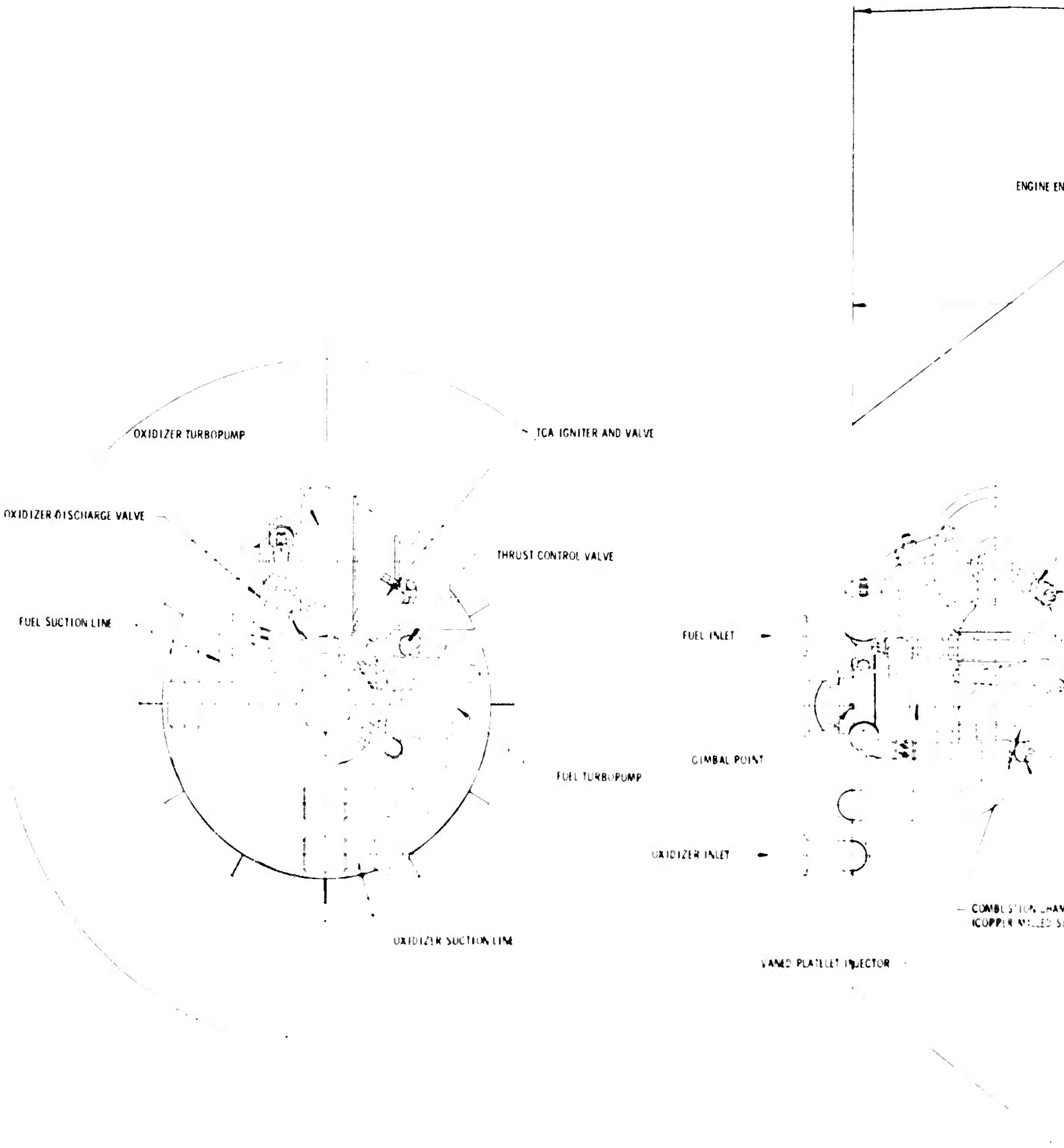
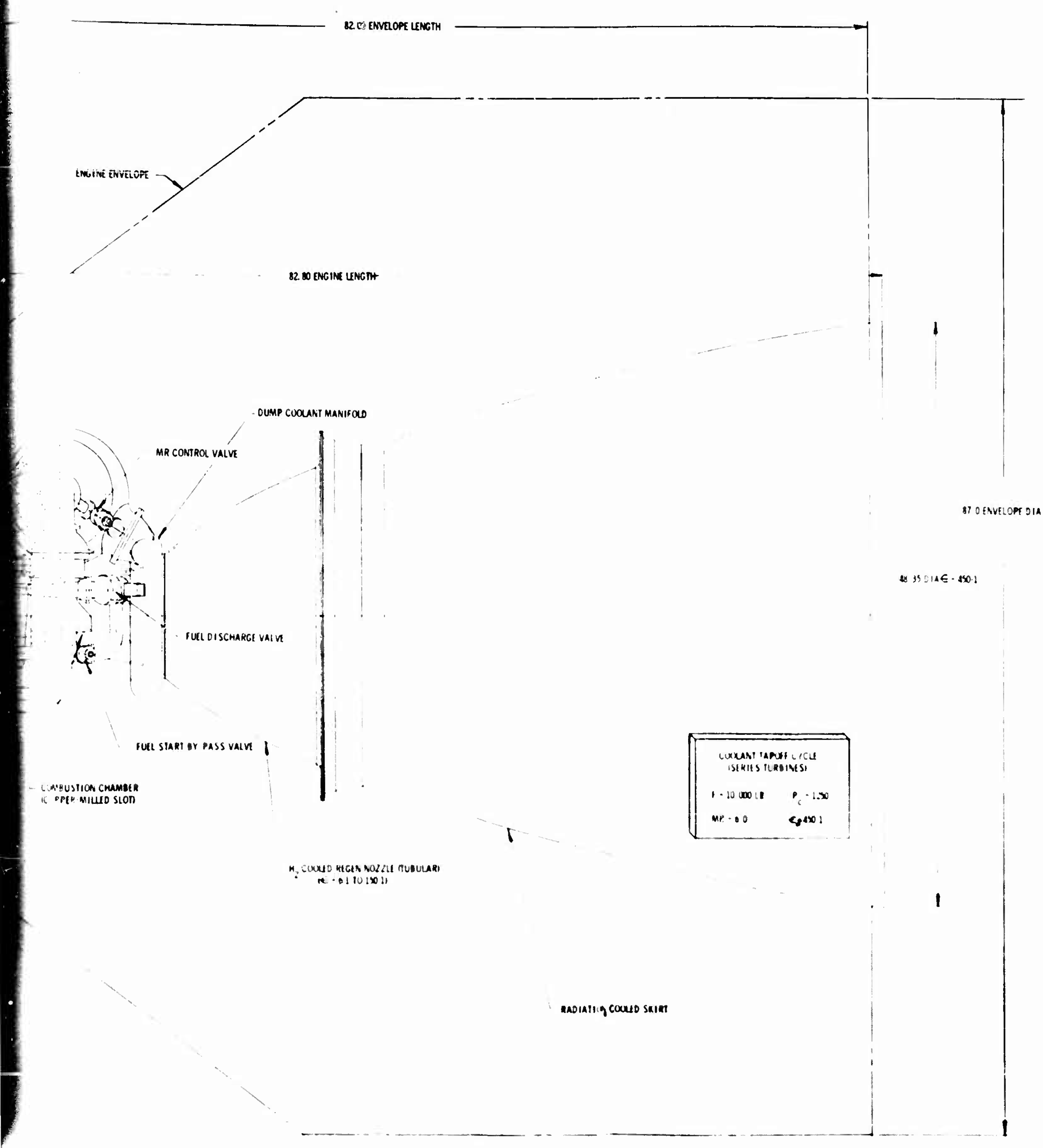


Figure 31. Configuration Dr.



ation Drawing, Coolant Bleed Cycle

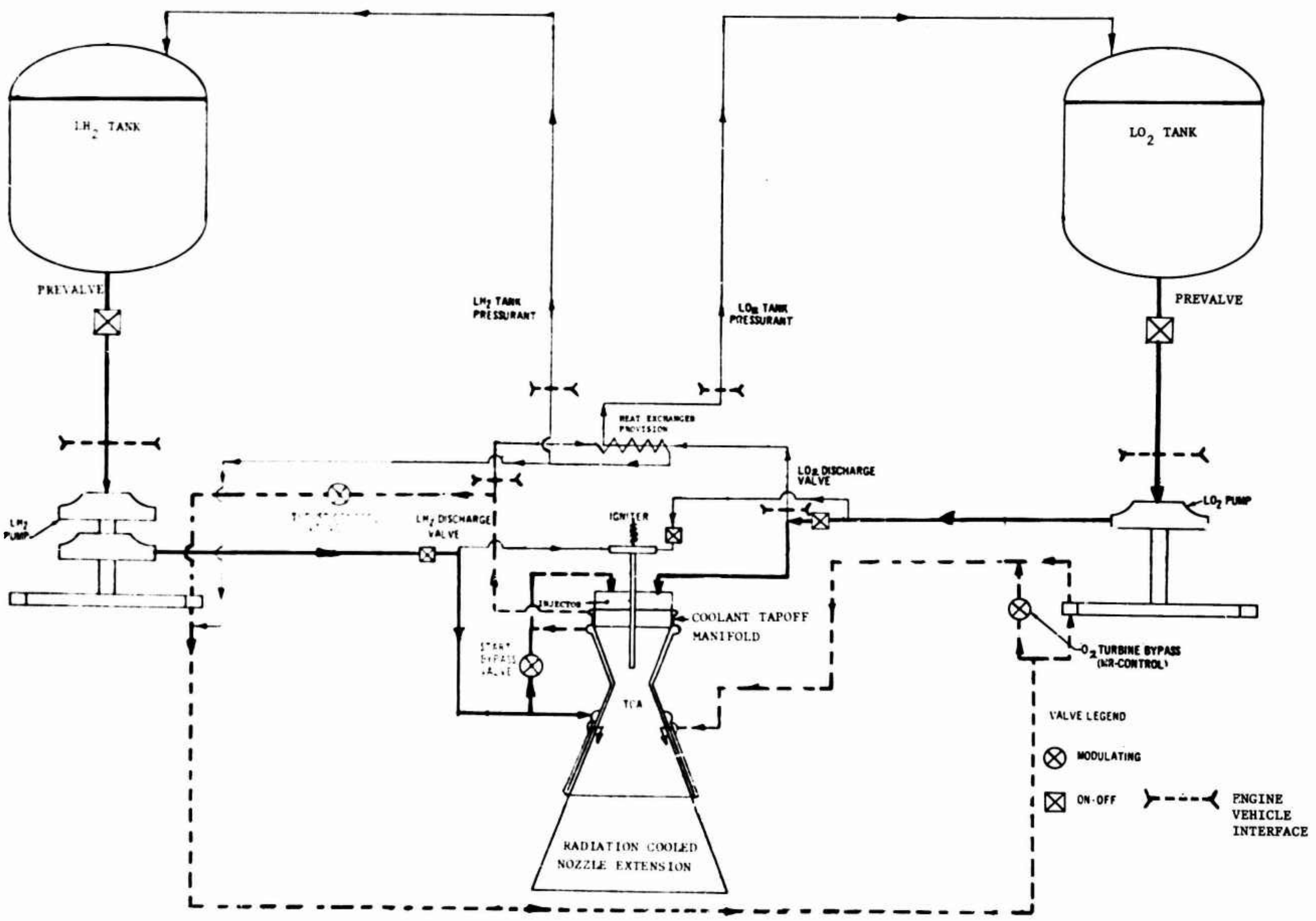


Figure 32. Engine Schematic, Coolant Bleed Cycle

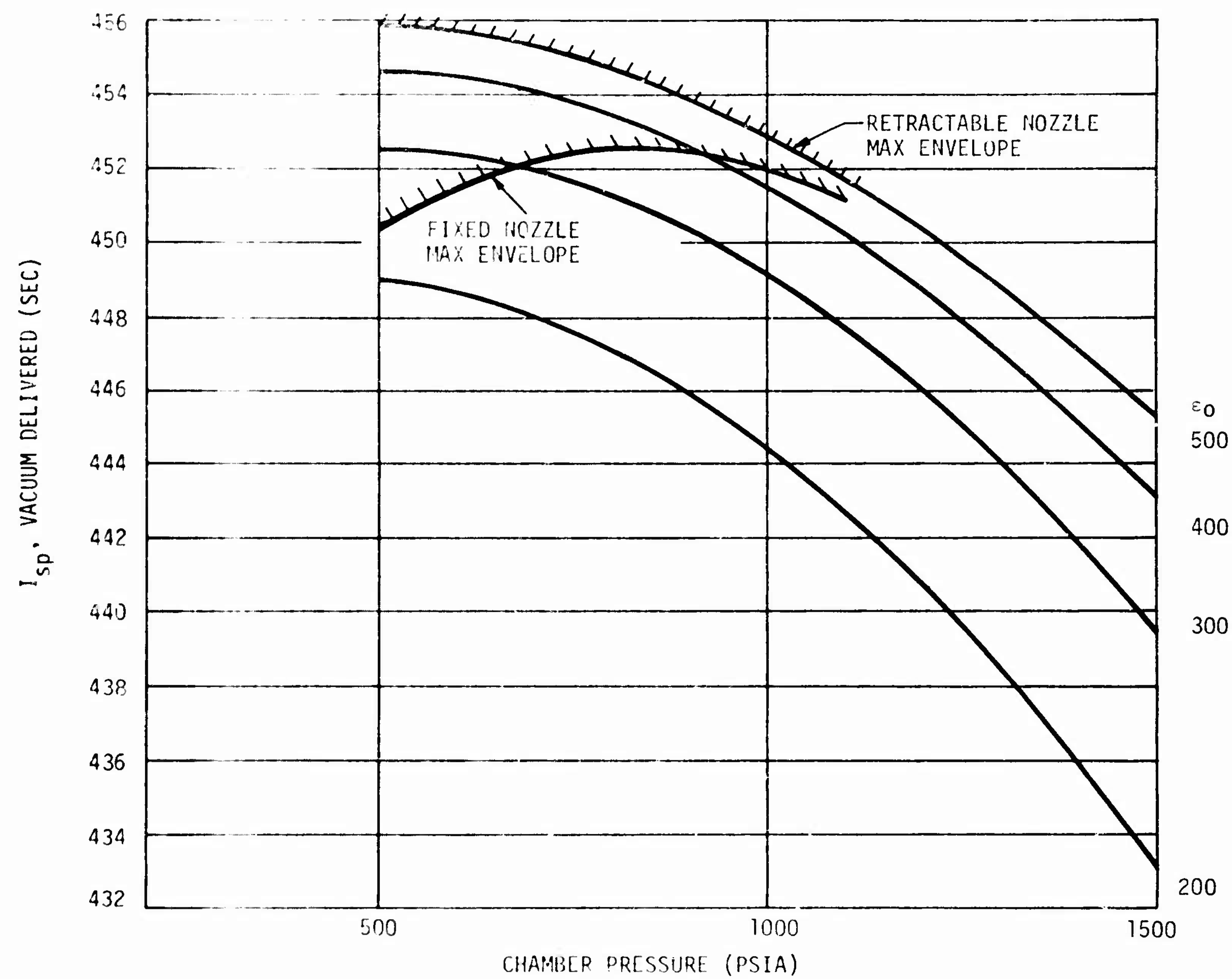


Figure 33. I_s vs P_c, MR = 6, F = 8K, Coolant Bleed Cycle

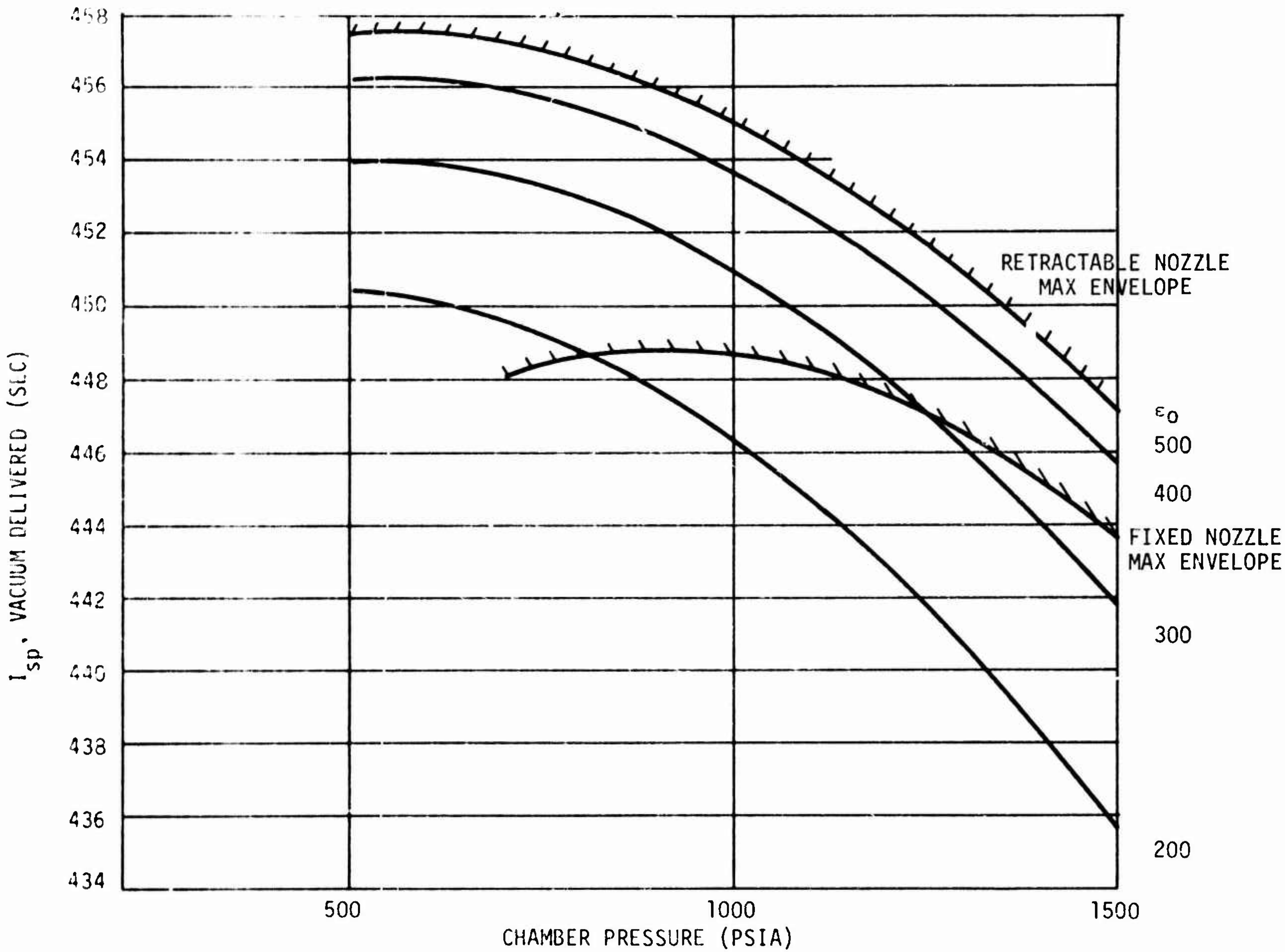


Figure 34. I_s vs P_c, MR = 6, F = 15K, Coolant Bleed Cycle

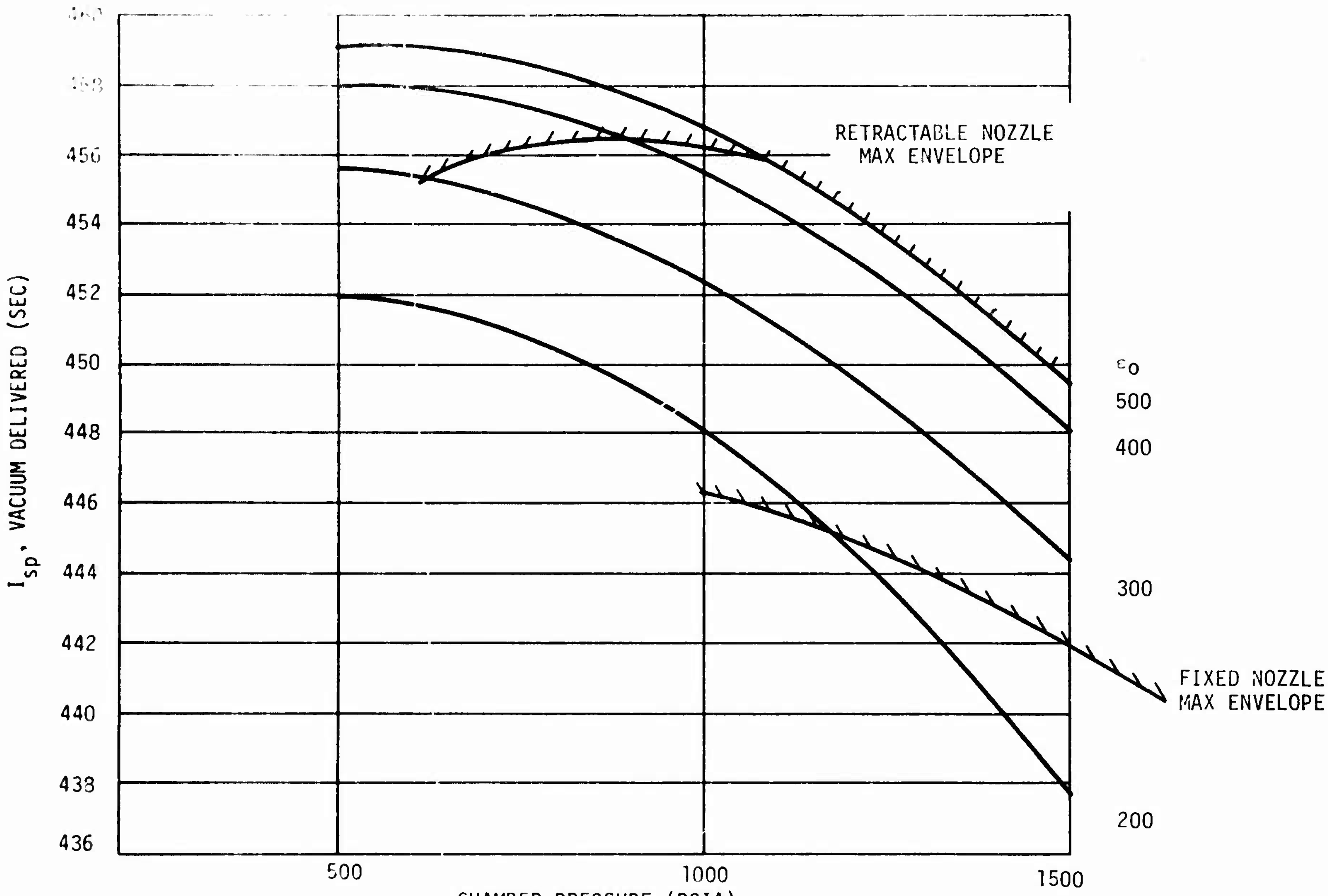


Figure 35. I_s vs P_c, MR = 6, F = 25K, Coolant Bleed Cycle

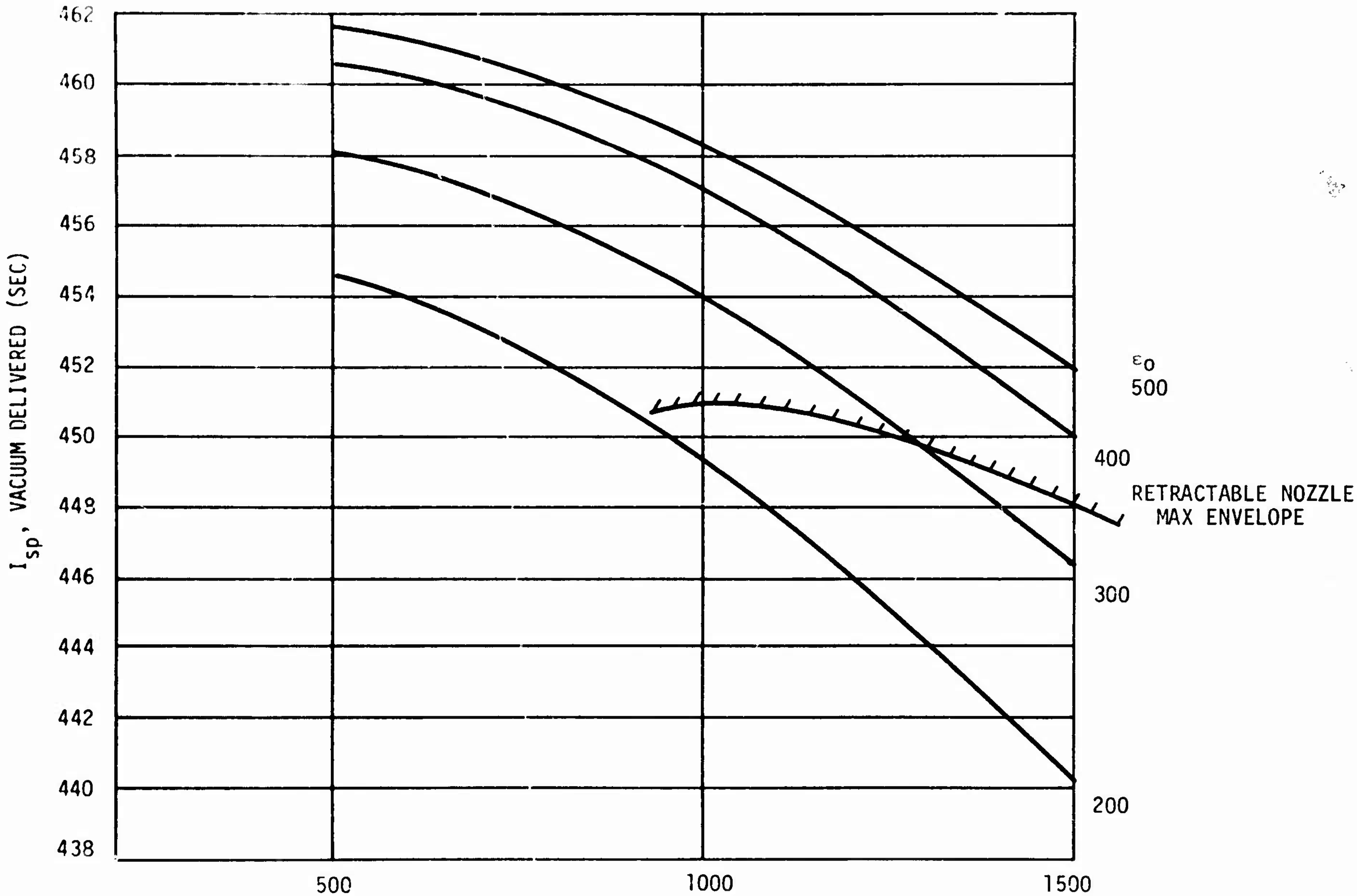


Figure 36. I_s vs P_c, MR = 6, F = 50K, Coolant Bleed Cycle

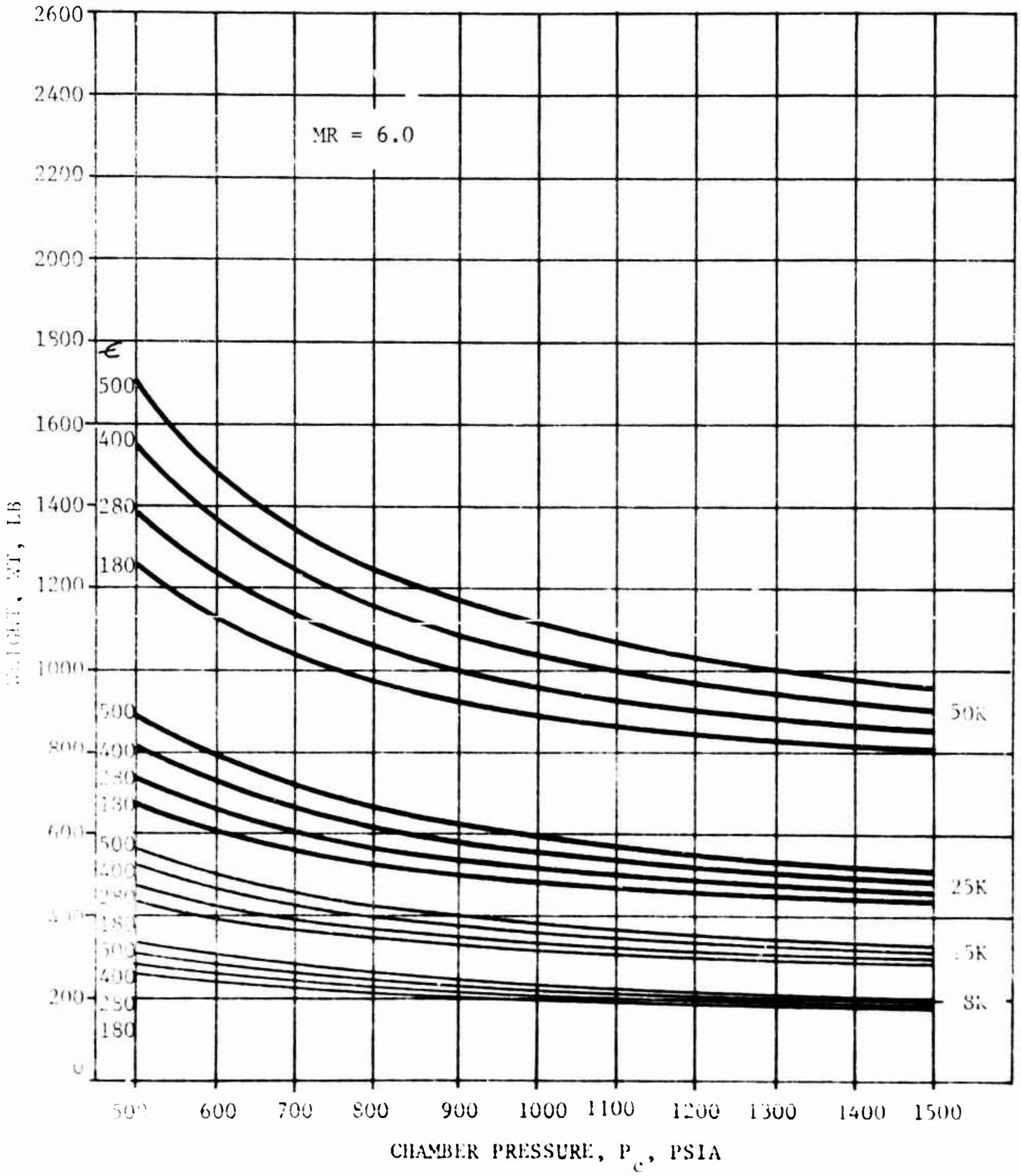


Figure 37. Engine Weight vs P_c, Fixed Nozzle, Coolant Bleed Cycle

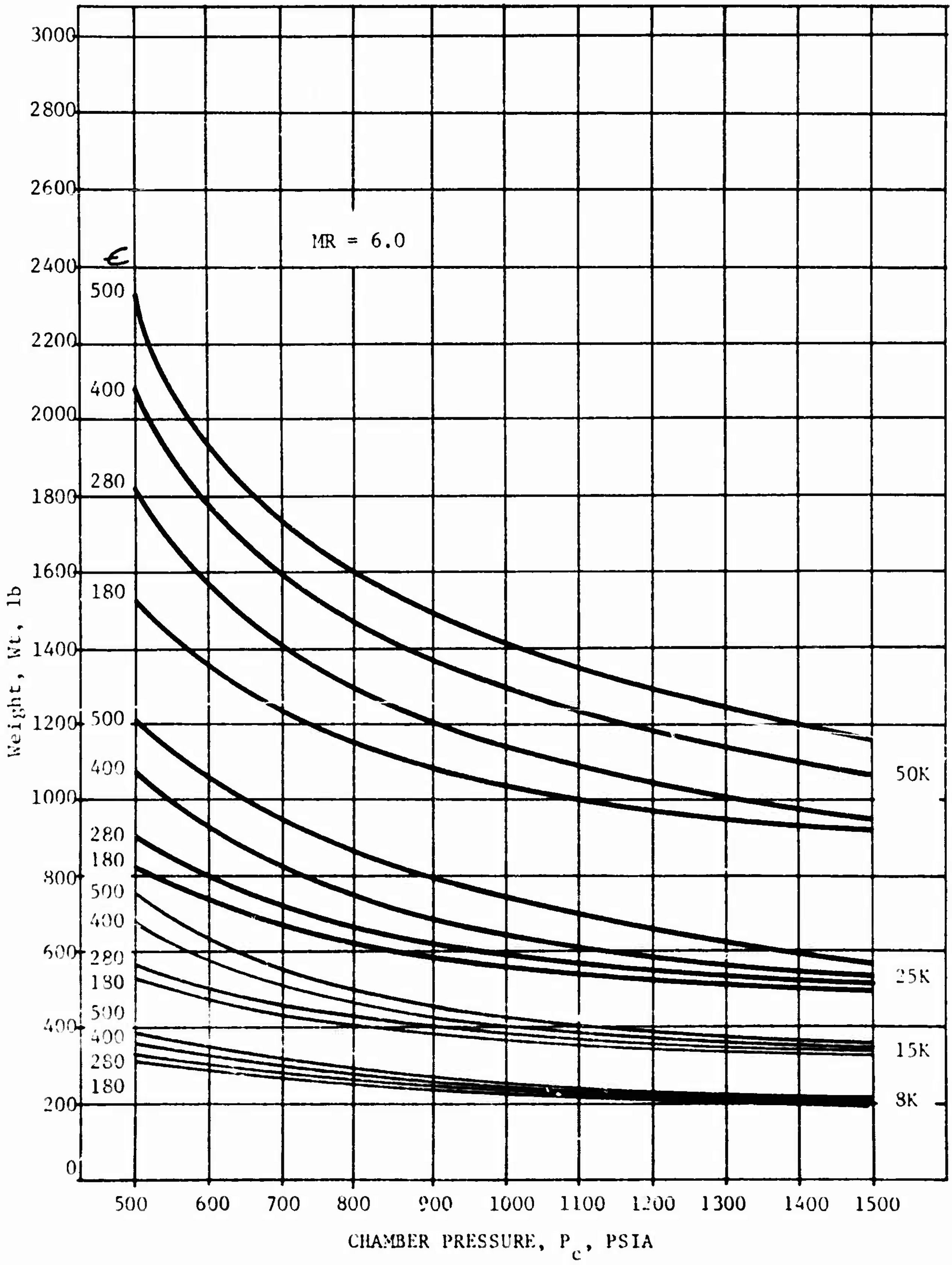


Figure 38. Engine Weight vs P_c, Minimum Weight Retractable Nozzle, Coolant Bleed Cycle

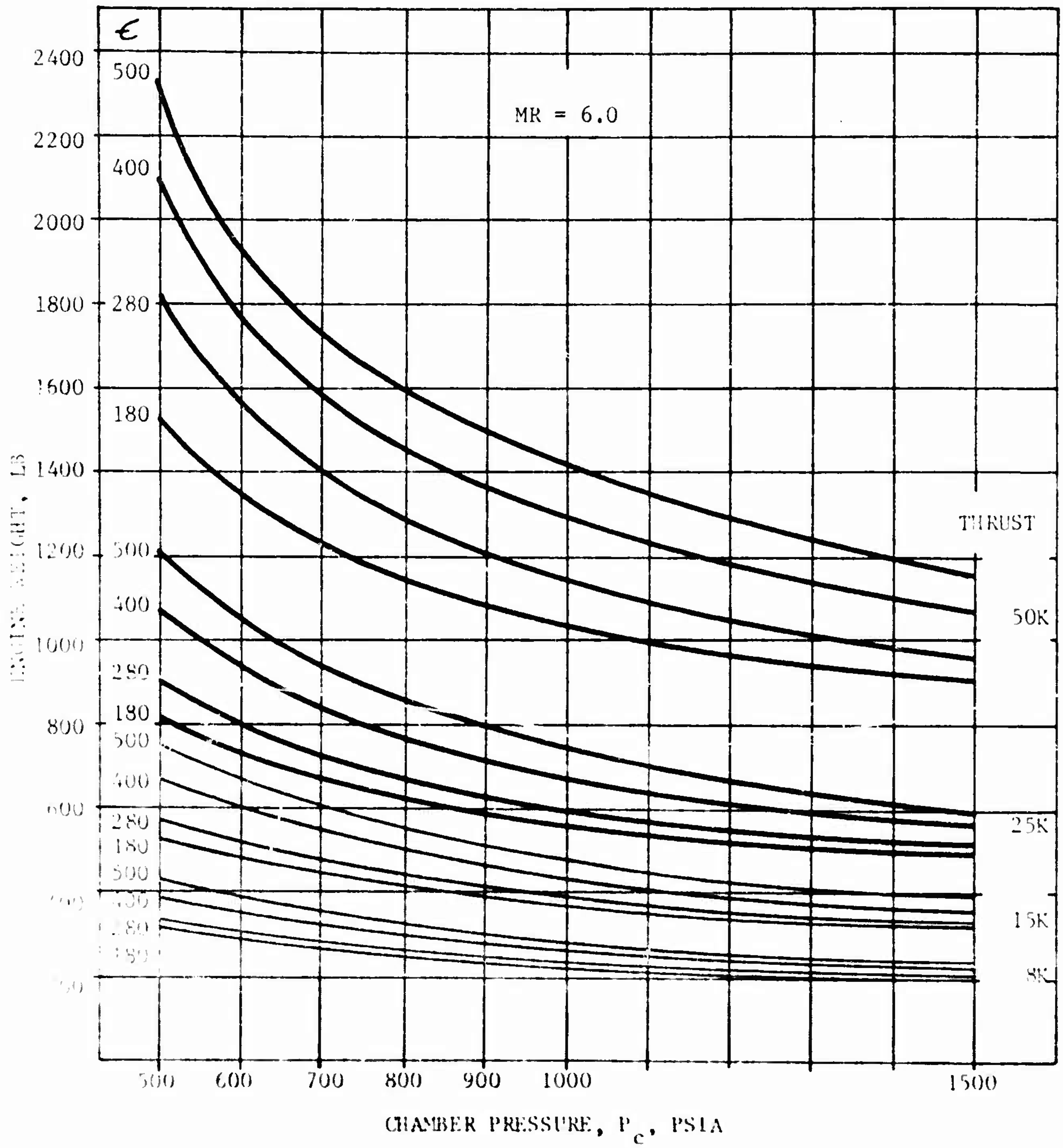


Figure 39. Engine Weight vs P_c, Minimum Length Retractable Nozzle, Coolant Bleed Cycle

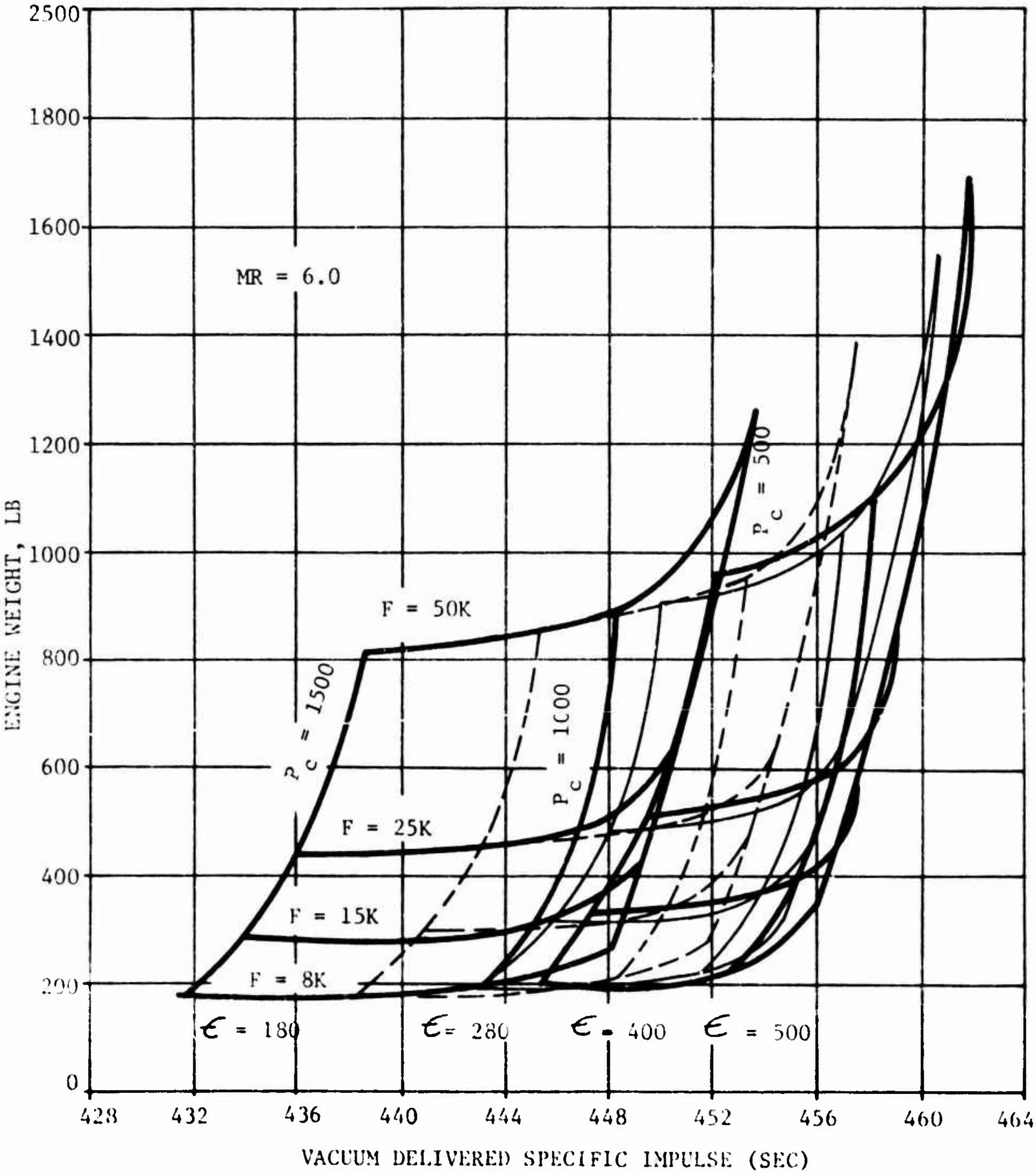


Figure 40. Engine Weight vs I_s, Fixed Nozzle, Coolant Bleed Cycle

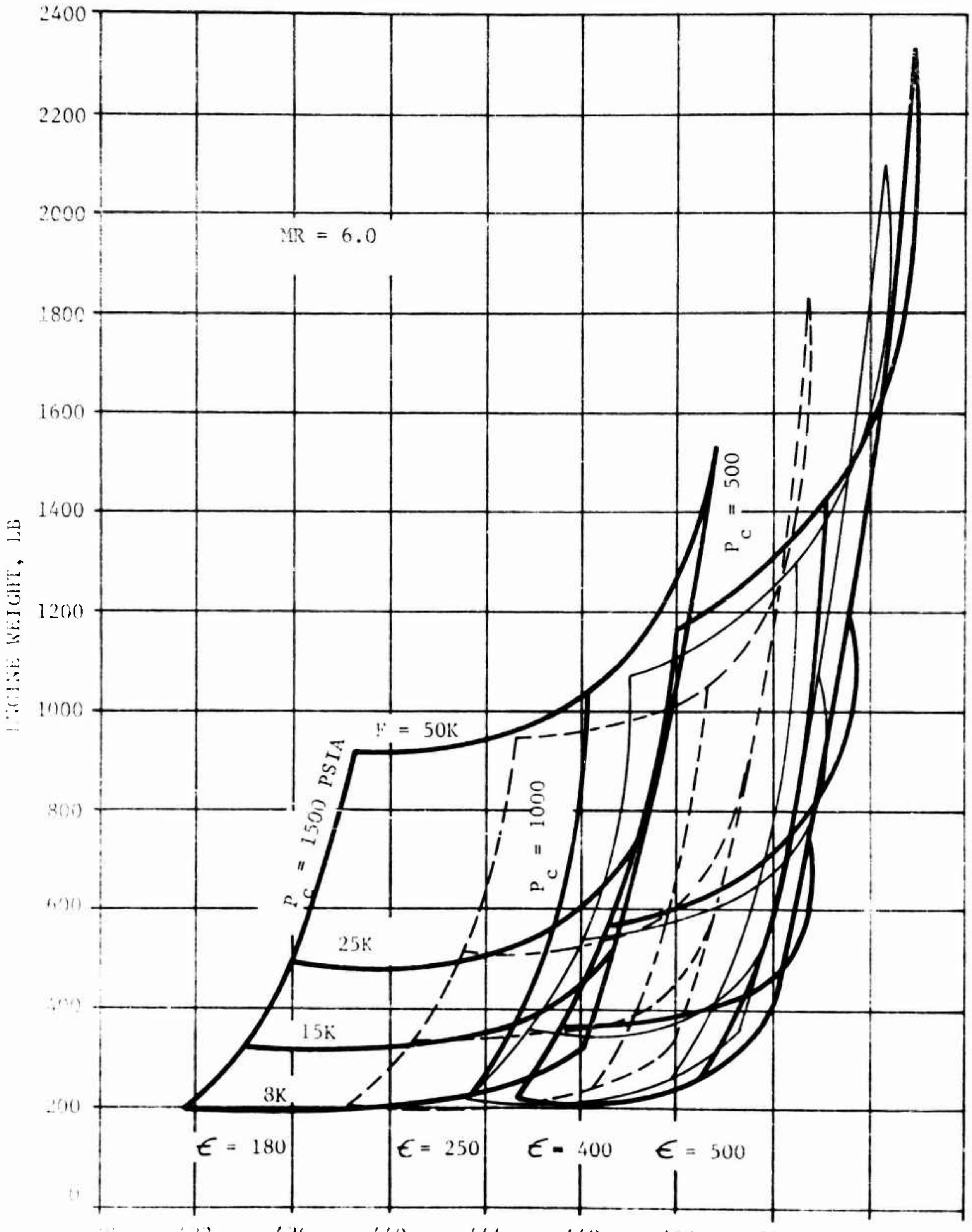


Figure 41. Engine Weight vs I_s, Minimum Weight Retractable Nozzle, Coolant Bleed Cycle

(4) Expander Cycle

This section includes the calculated weight of the expander cycle for mixture ratio = 6.0

The engine performance is equal to the thrust chamber performance. Since the staged combustion cycle performance is identical, the expander cycle performance is not shown separately.

All engine data shown are for a 14-in. thrust chamber length.

Figure 42 is the configuration drawing and Figure 43 the schematic. Figures 44, 45 and 46 are weight data.

The interaction of engine weight and performance are shown in Figures 47 and 48. All weights include low speed pumps.

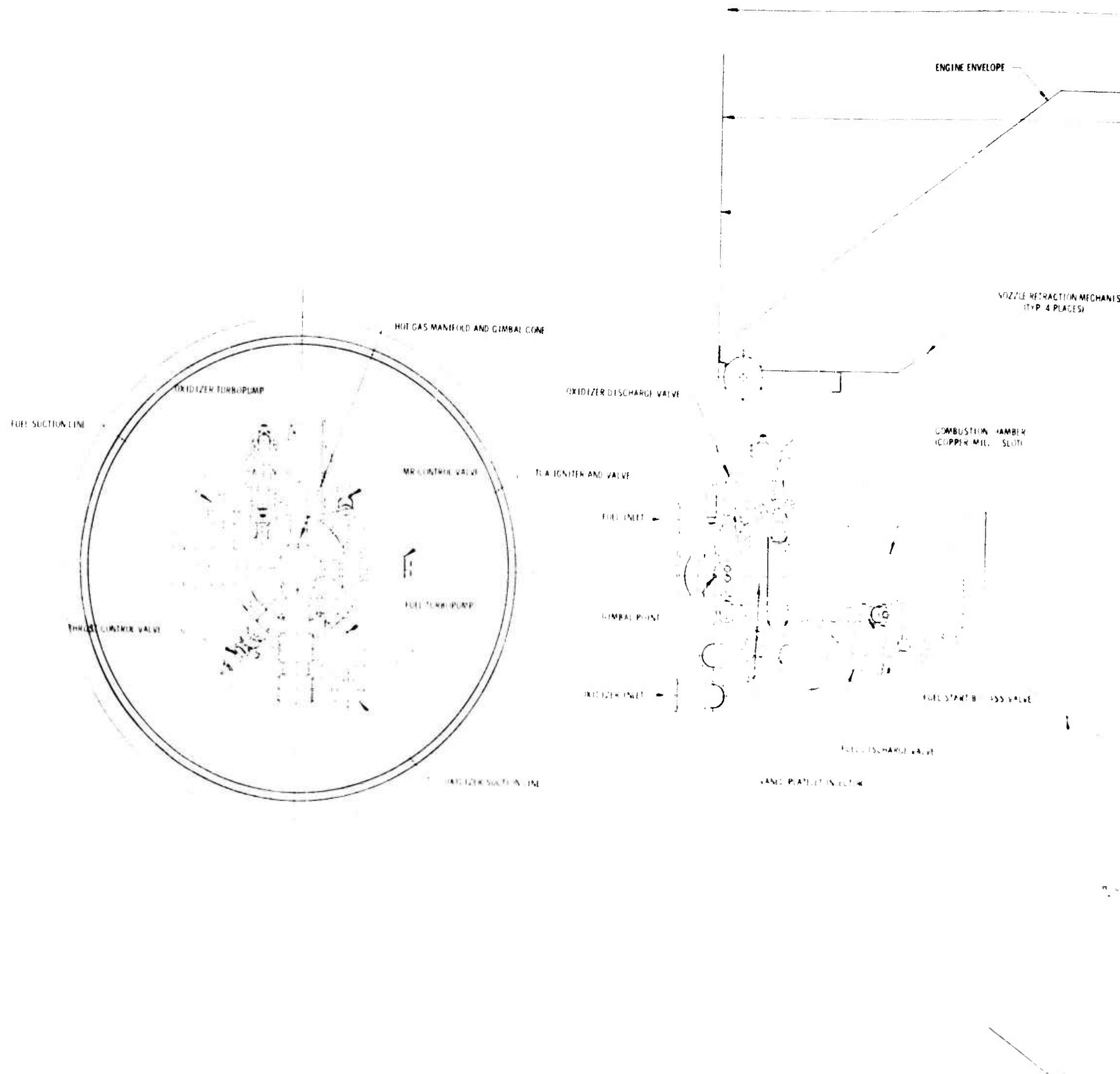
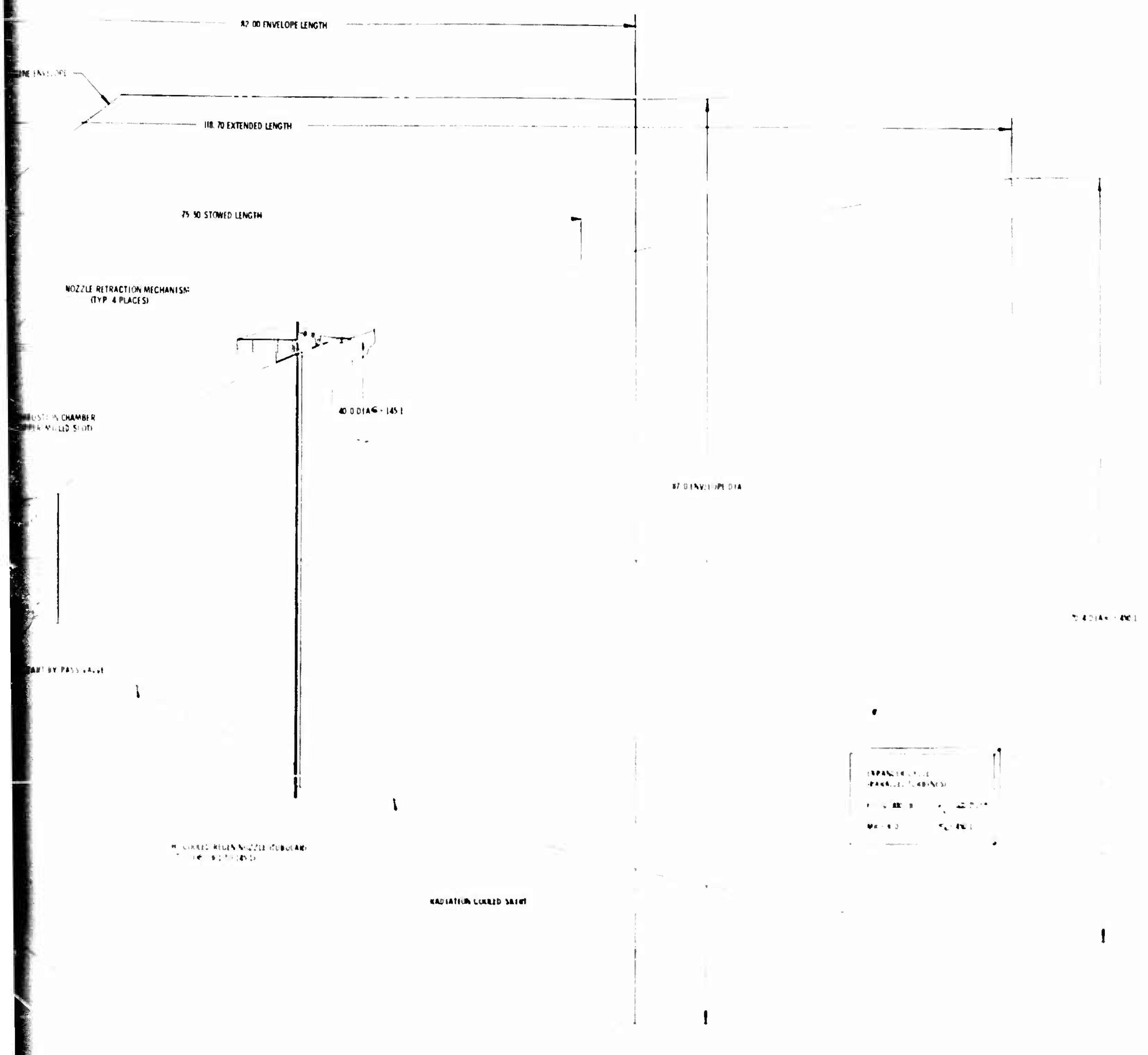


Figure 42. Configuration Drawi



Configuration Drawing, Expander Cycle

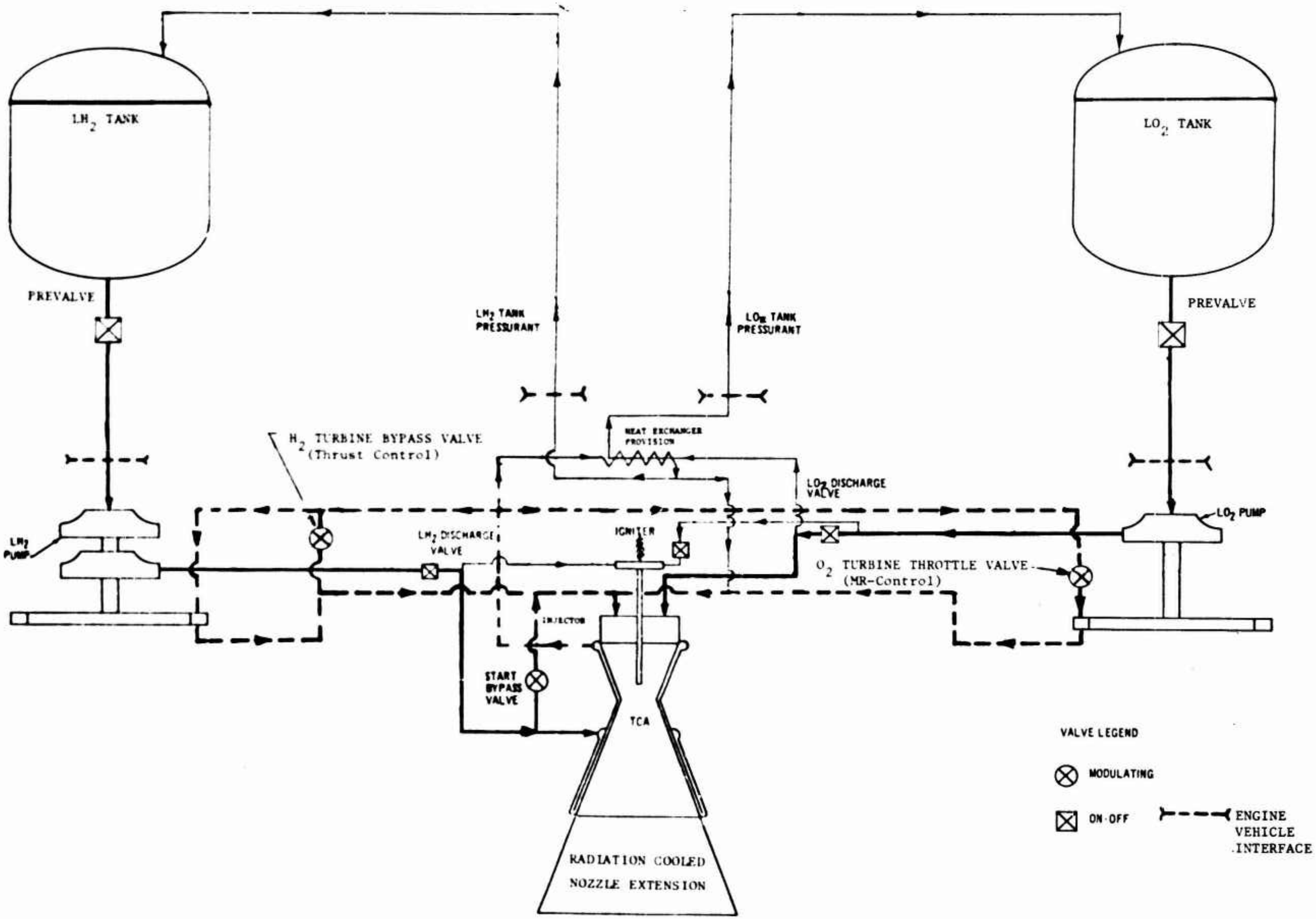


Figure 43. Engine Schematic, Expander Cycle

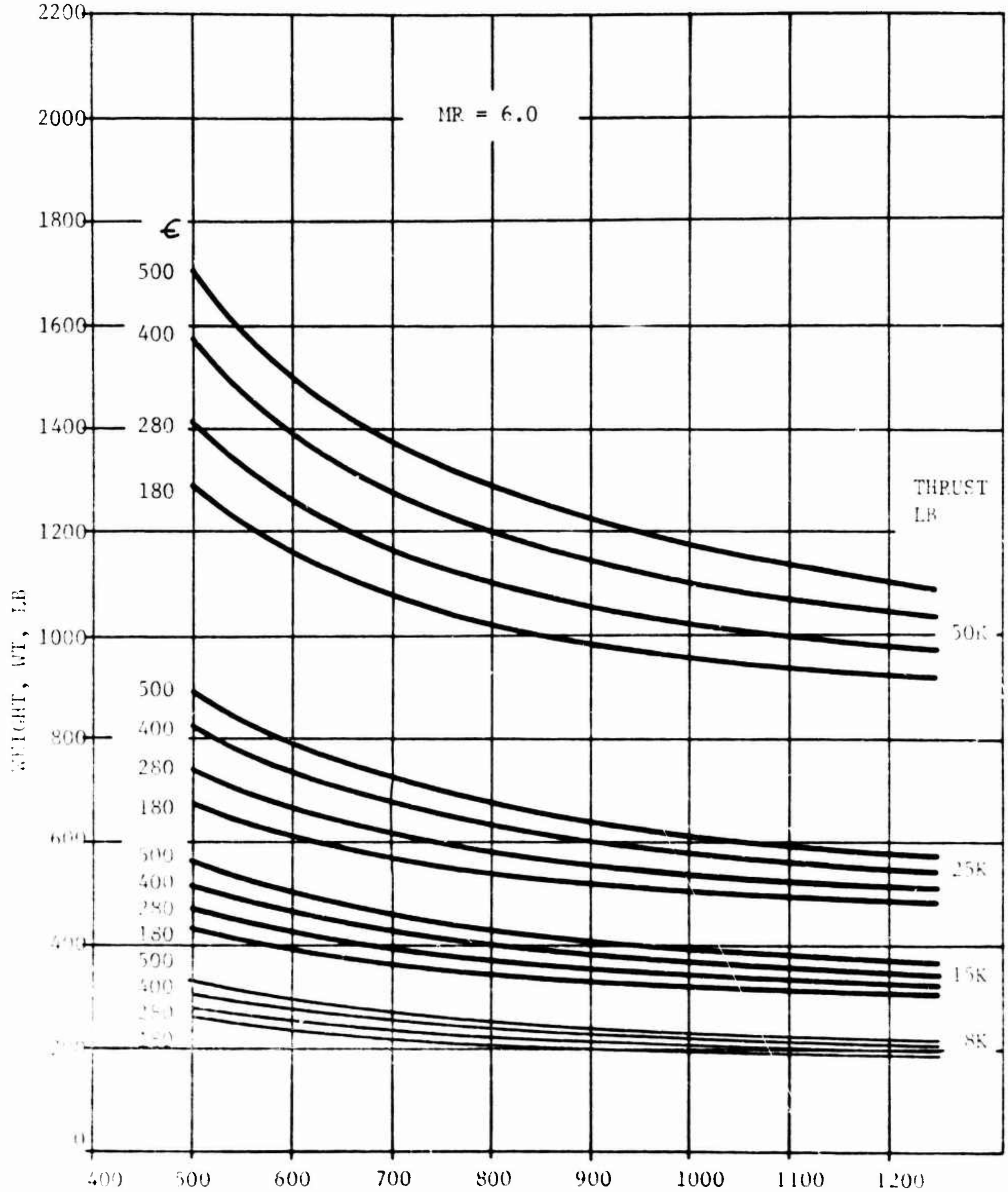


Figure 44. Engine Weight vs P_c, Fixed Nozzle, Expander Cycle

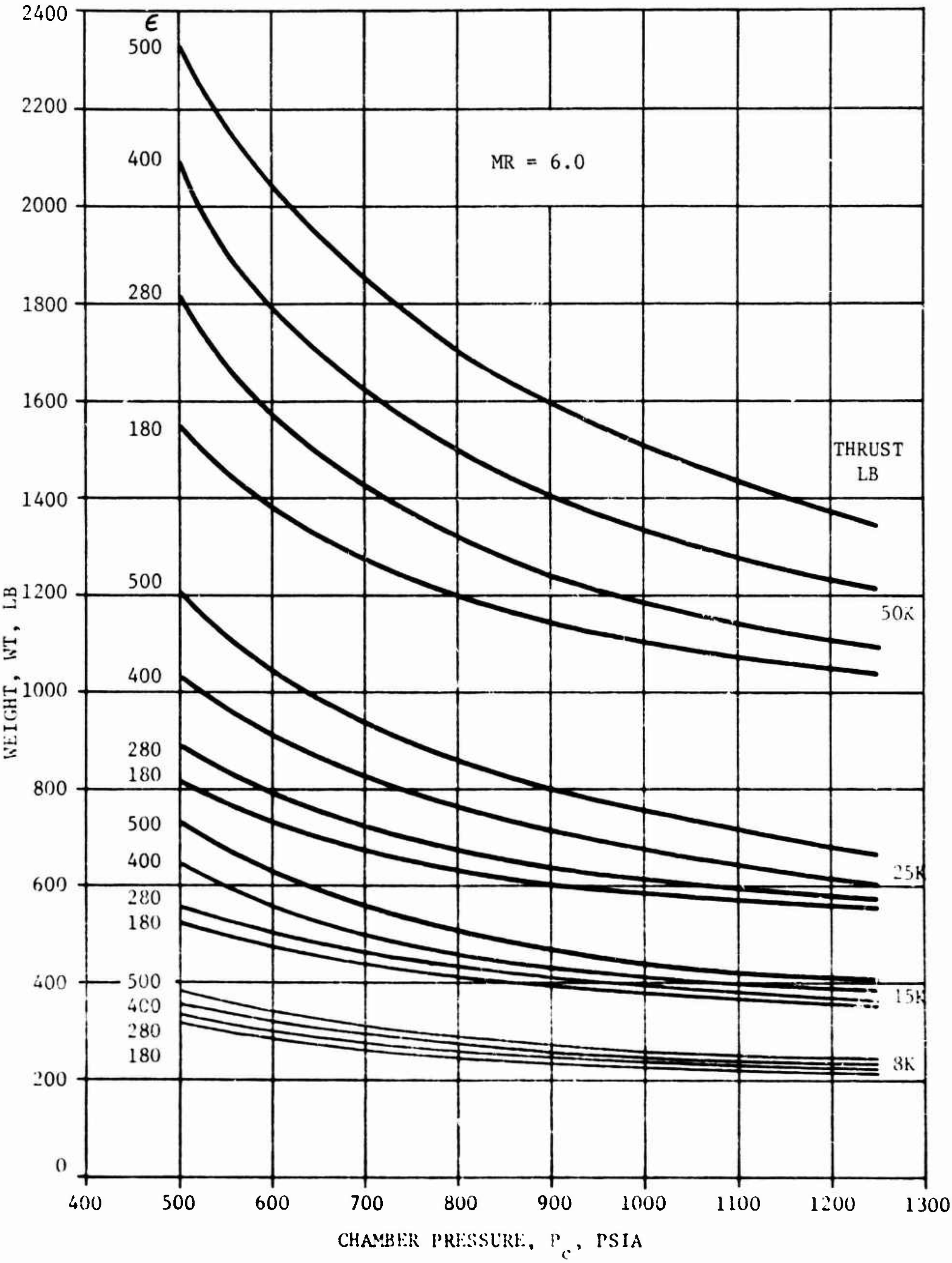


Figure 45. Engine Weight vs P_c, Minimum Weight Retractable Nozzle, Expander Cycle

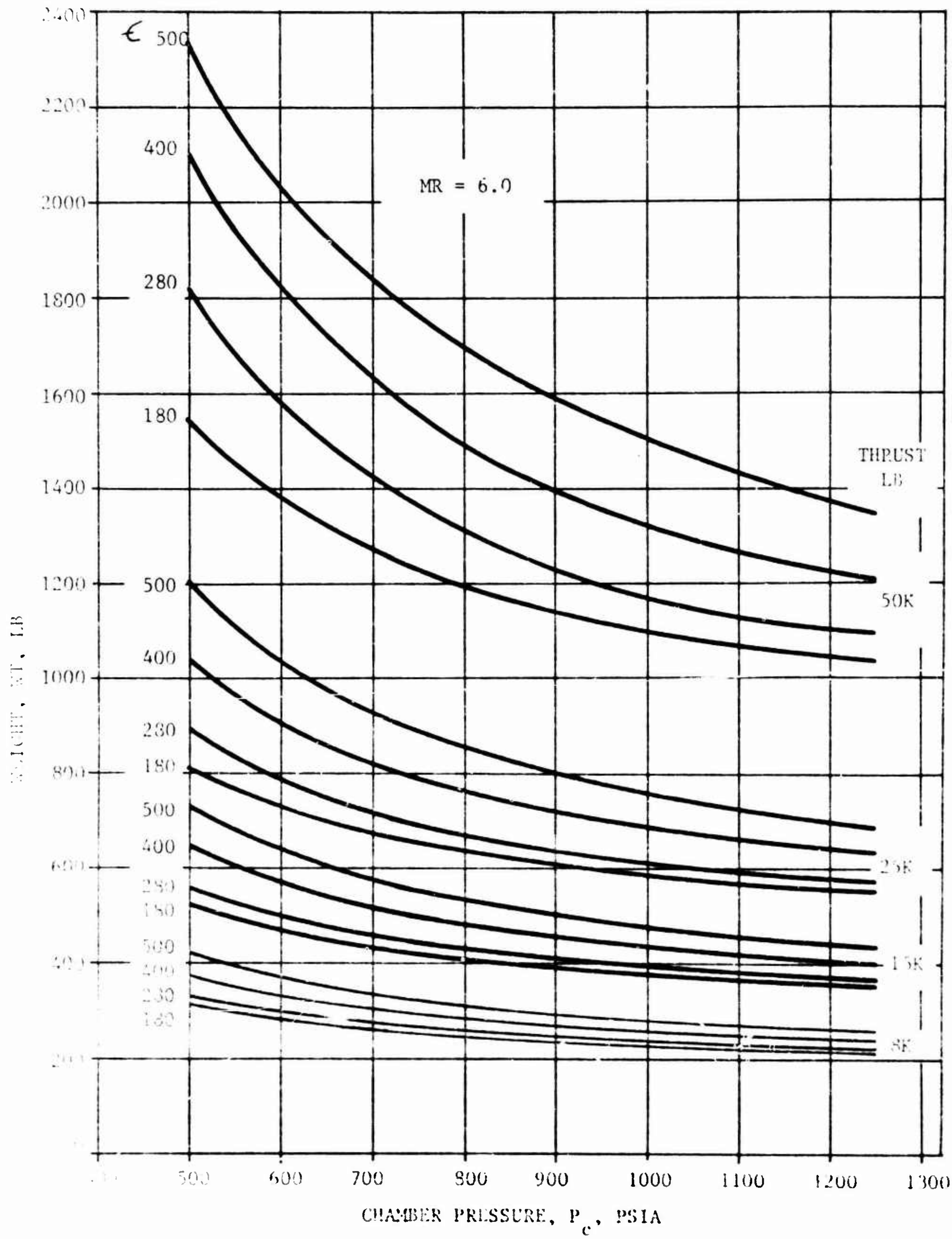


Figure 46. Engine Weight vs P_c, Minimum Length Retractable Nozzle, Expander Cycle

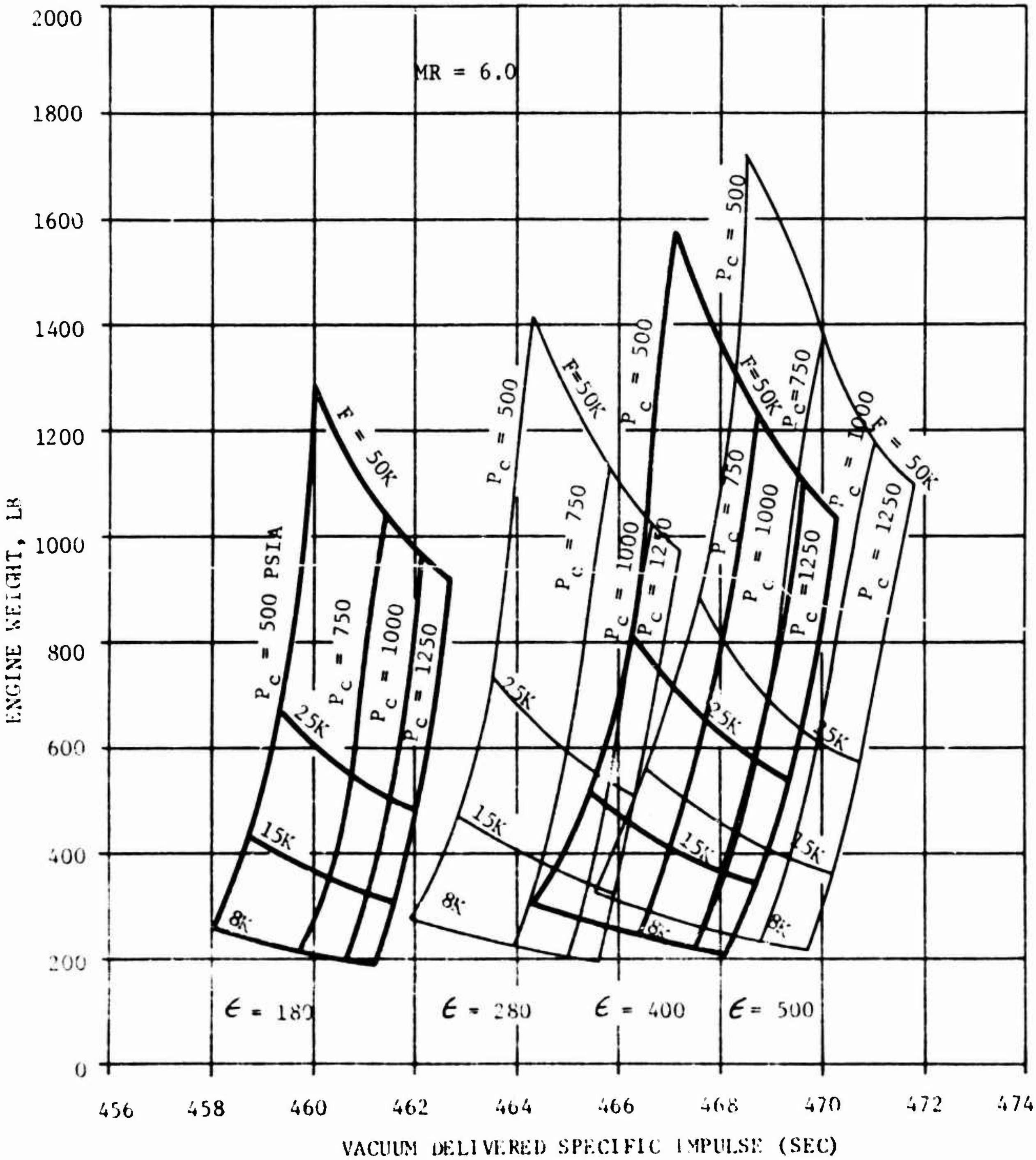


Figure 47. Engine Weight vs I_s, Fixed Nozzle, Expander Cycle

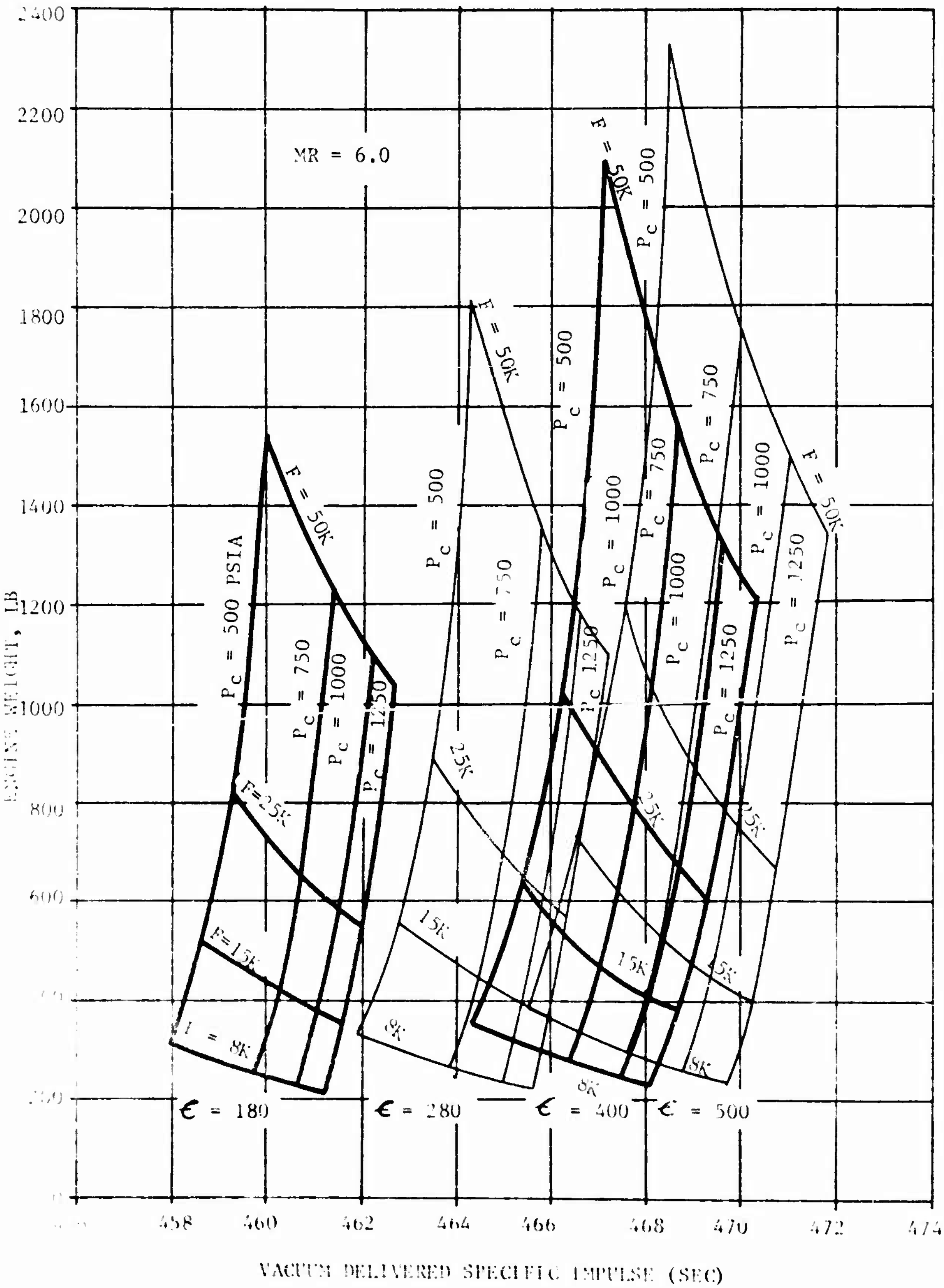


Figure 48. Engine Weight vs I_s, Minimum Weight Retractable Nozzle, Expander Cycle

(5) Staged Combustion Cycle

This engine cycle is the highest performing cycle over the whole thrust range of 8K to 50K lb.

Figures 49 and 50 are the configuration and schematic drawings and Figures 51 through 57 are the appropriate performance and weight data. Figures 58 and 59 are the interaction charts. All weights shown include low speed pumps.

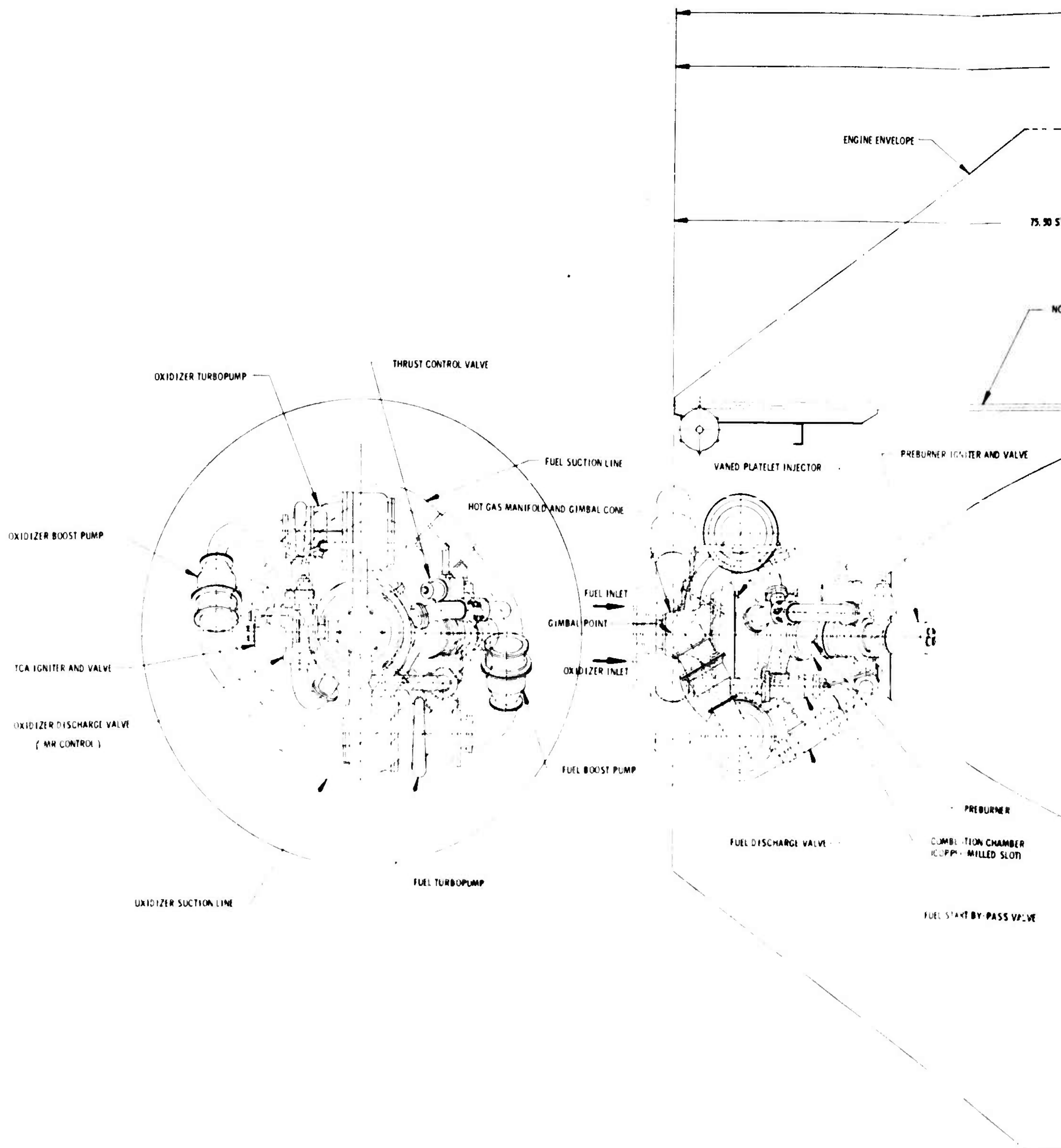
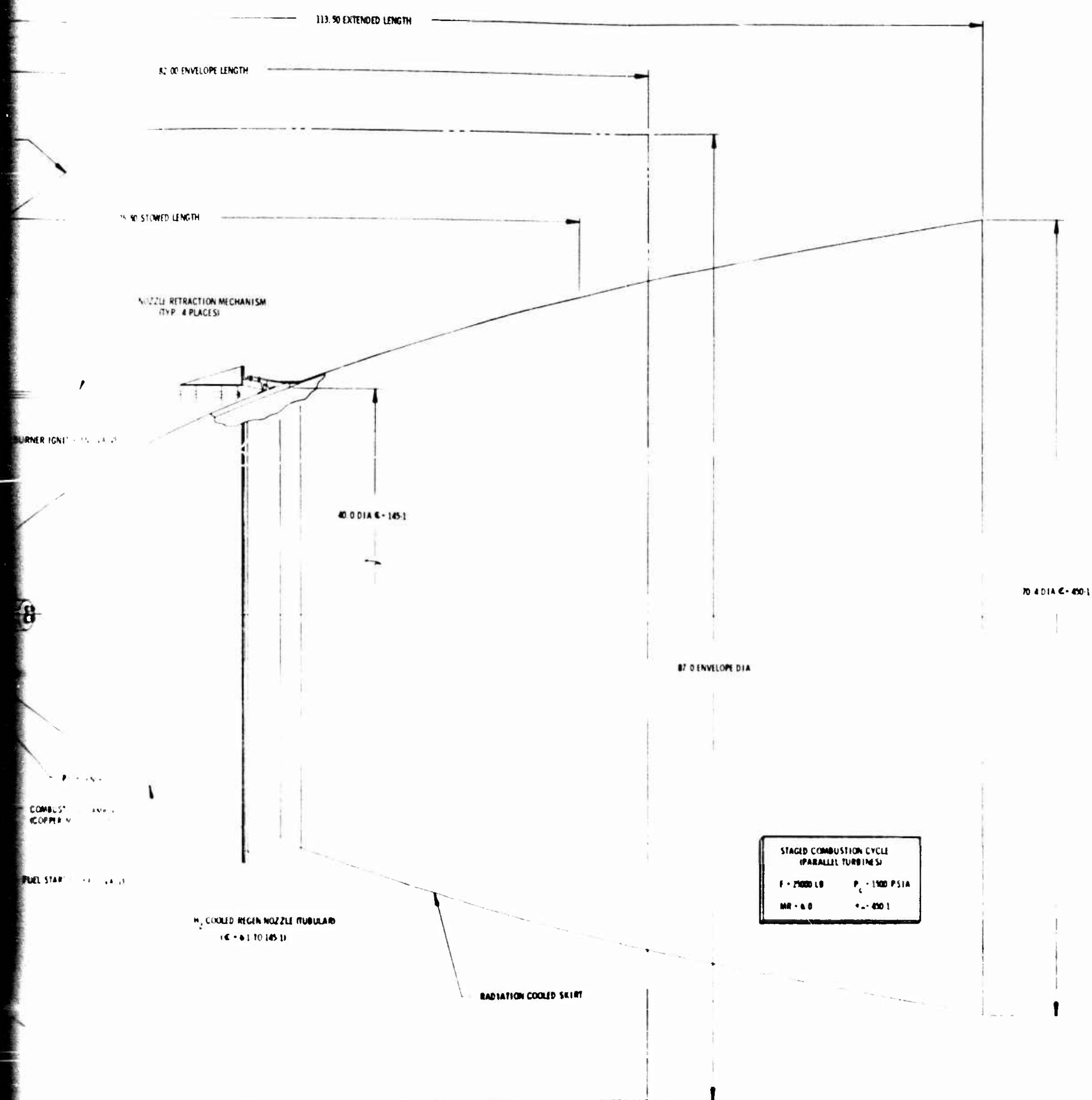


Figure 49. Configuration Drawing



Staged Combustion Cycle

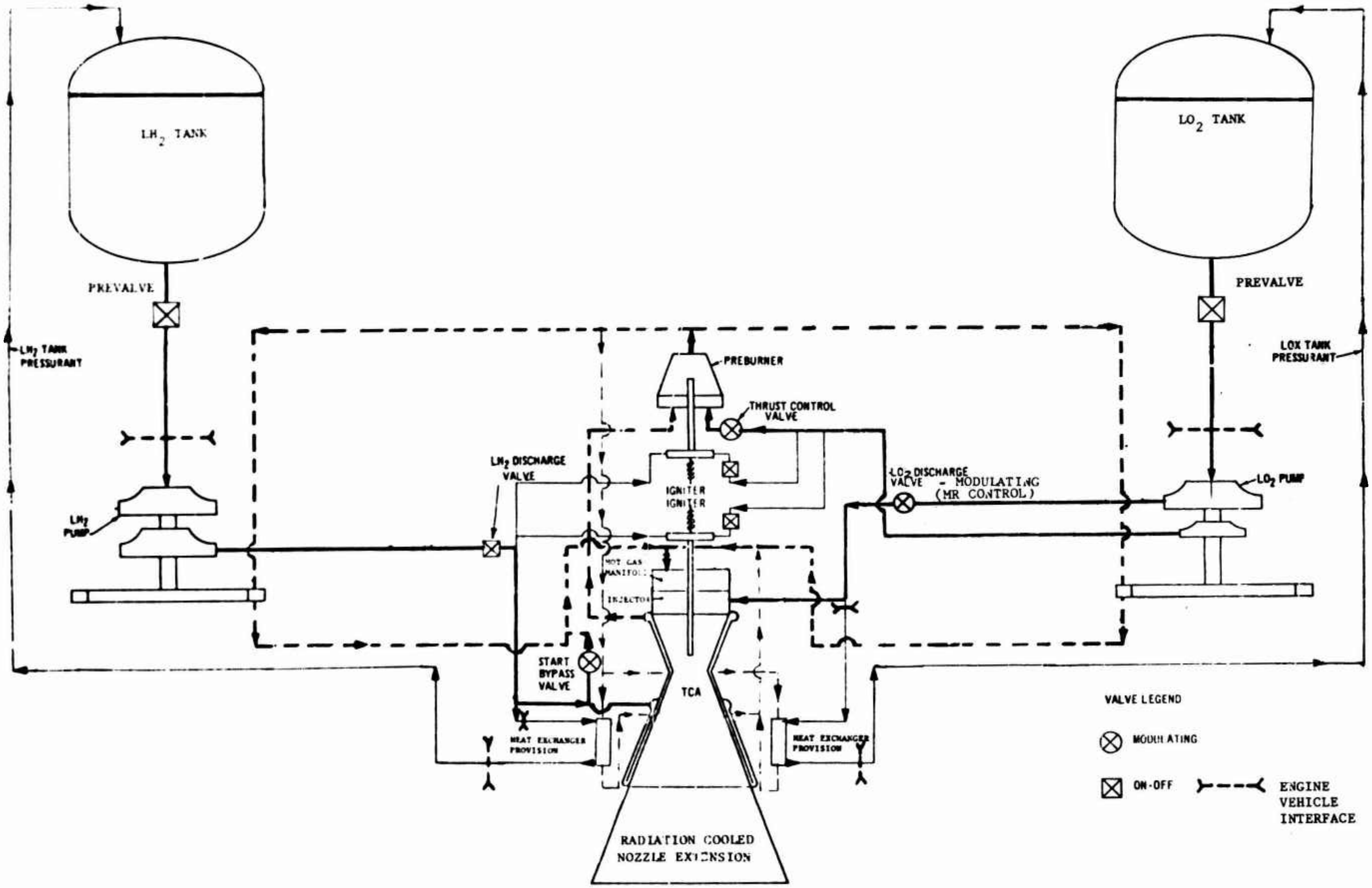


Figure 50. Engine Schematic, Staged Combustion Cycle

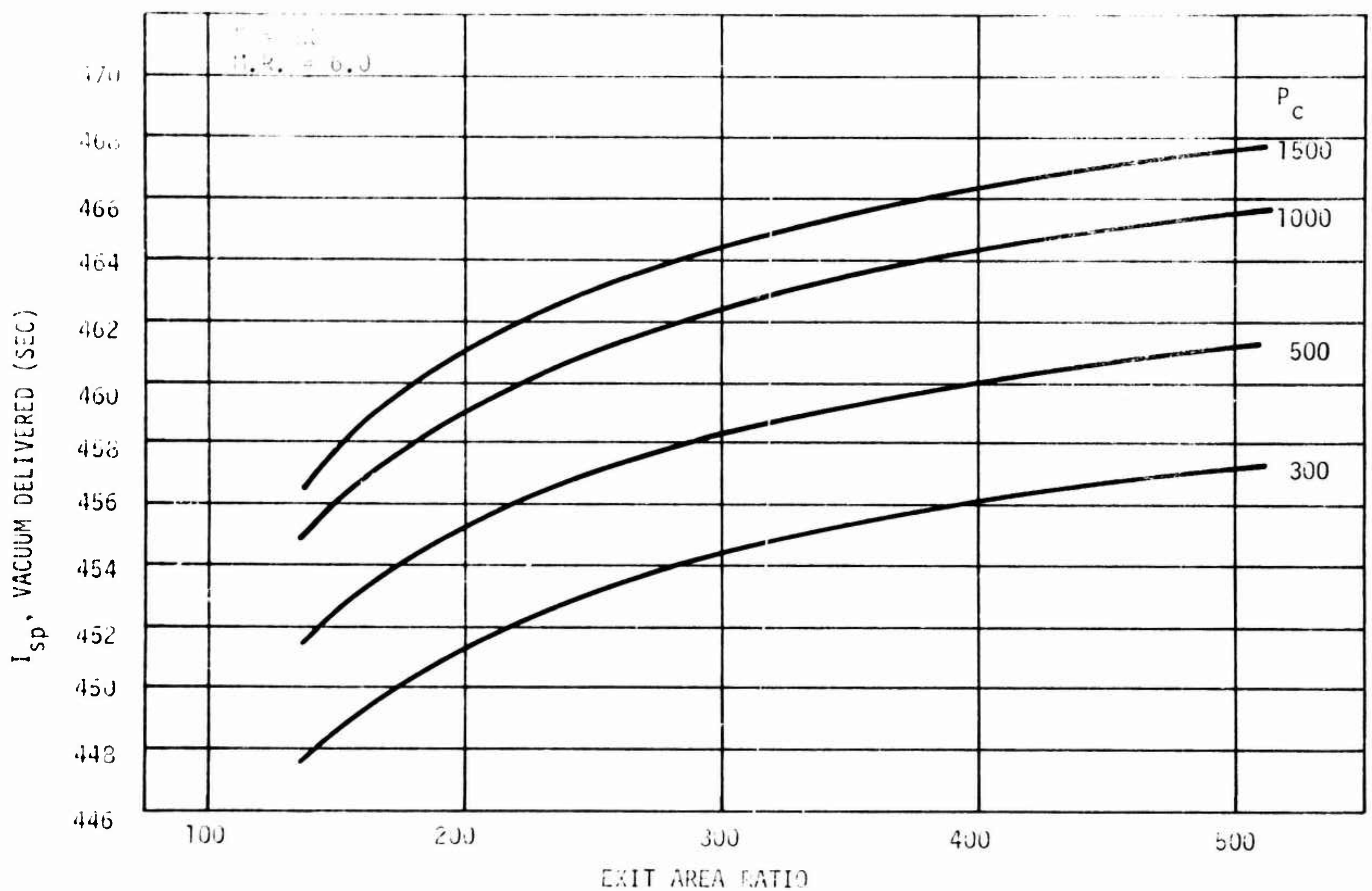


Figure 51. I_s vs Exit Area Ratio, MR = 6, F = 8K, Staged Combustion Cycle

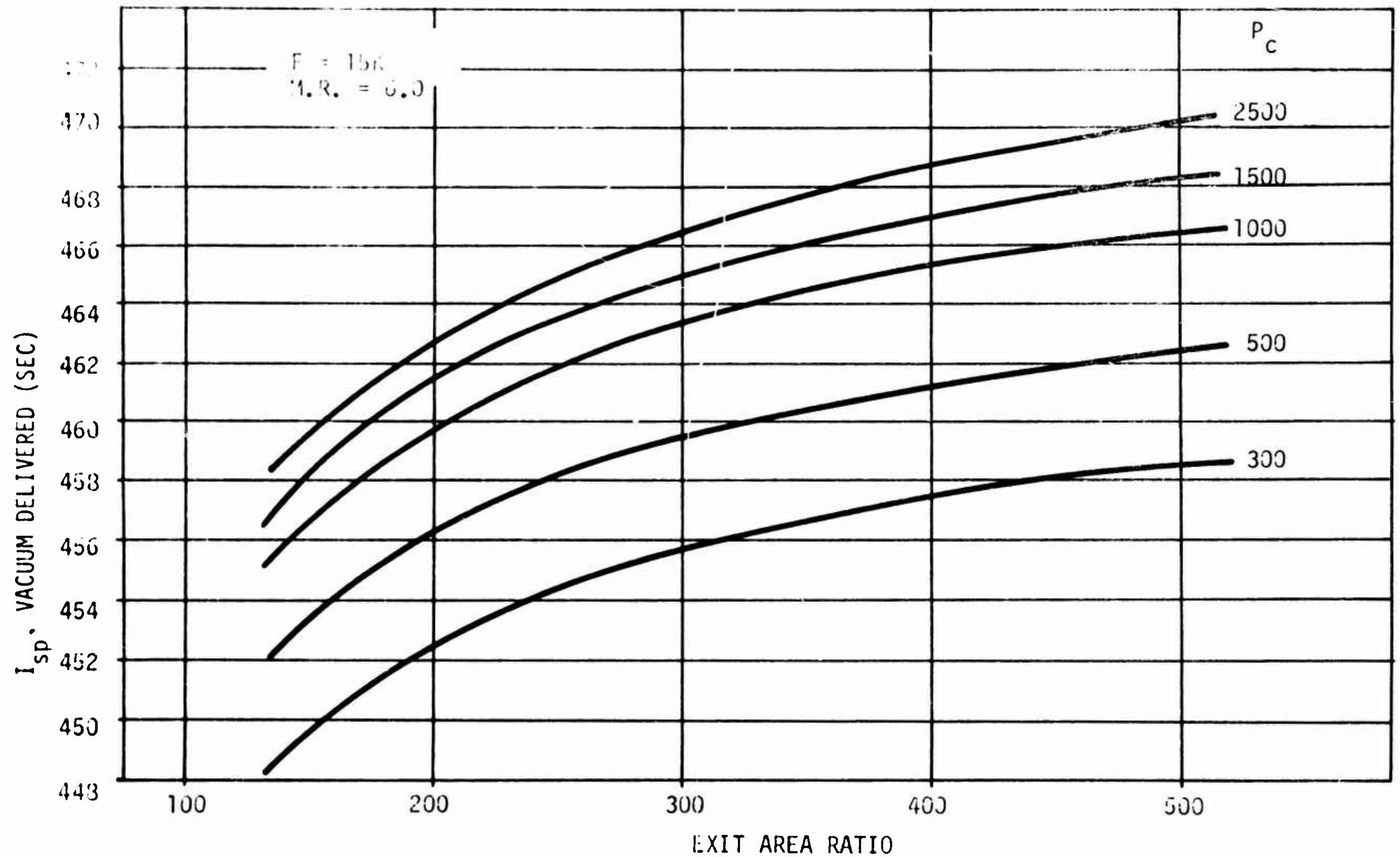


Figure 52. I_s vs Exit Area Ratio, MR = 6, F = 15K, Staged Combustion Cycle

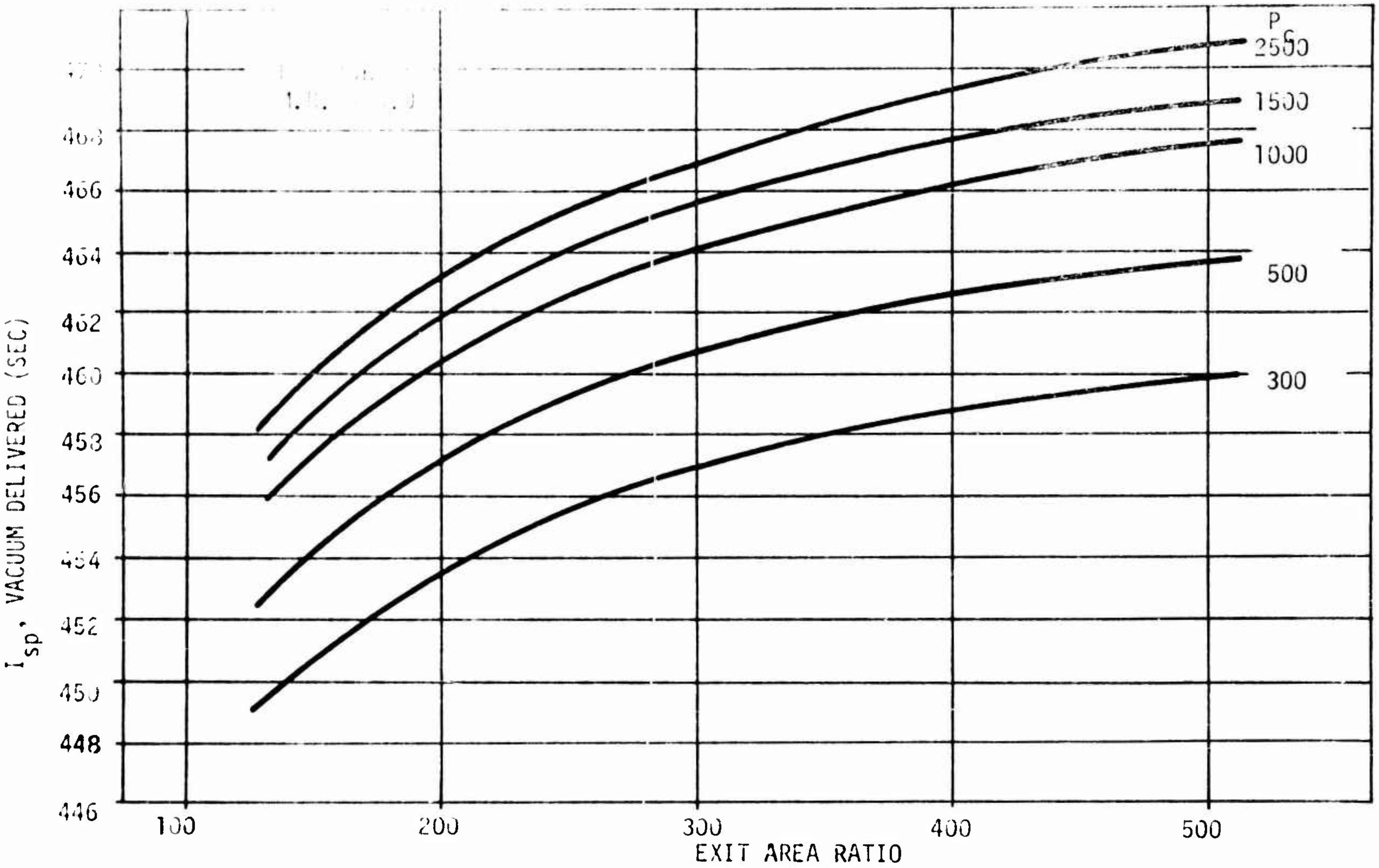


Figure 53. I_s vs Exit Area Ratio, MR = 6, F = 25K, Staged Combustion Cycle

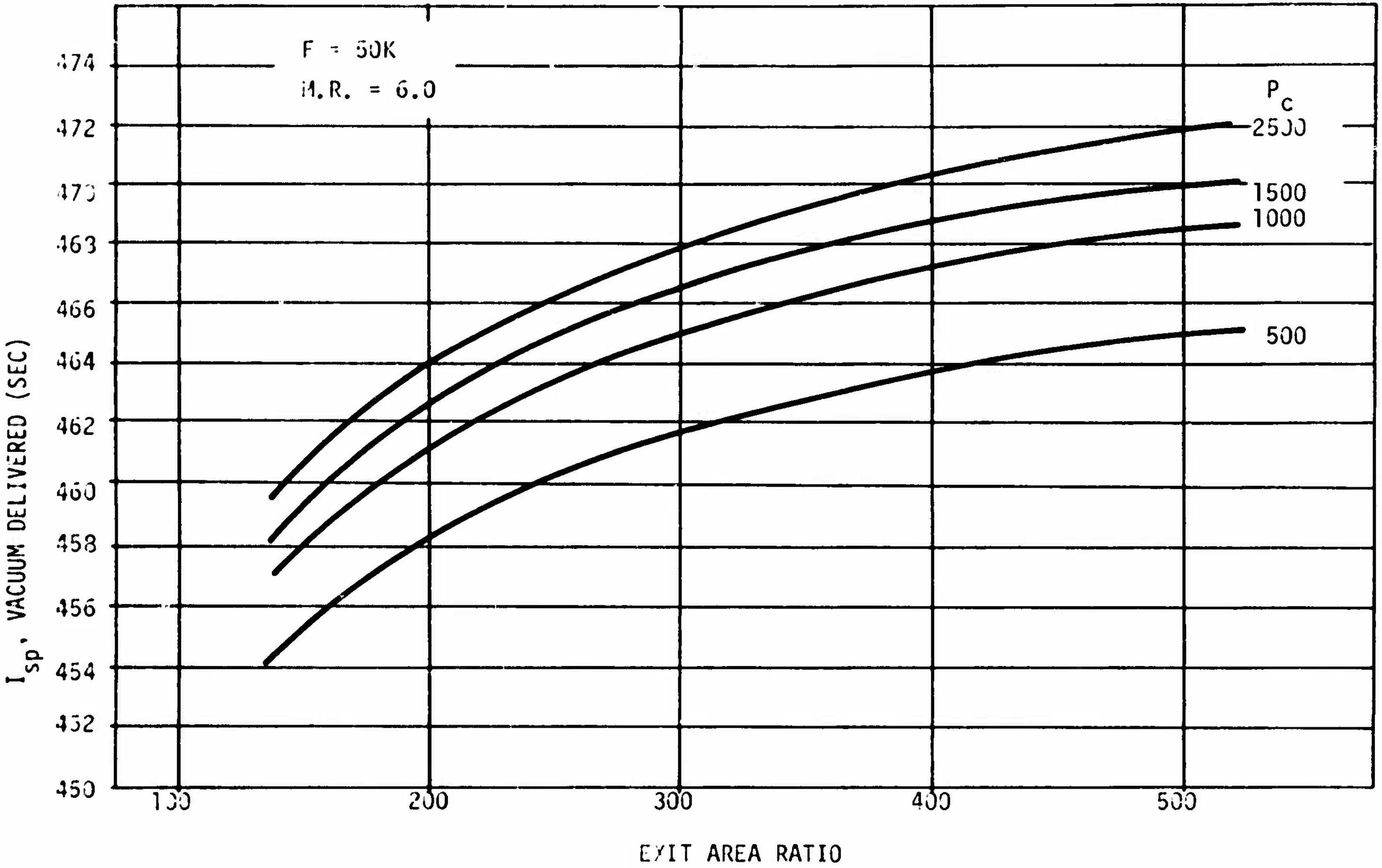


Figure 54. I_s vs Exit Area Ratio, MR = 6, F = 50K, Staged Combustion Cycle

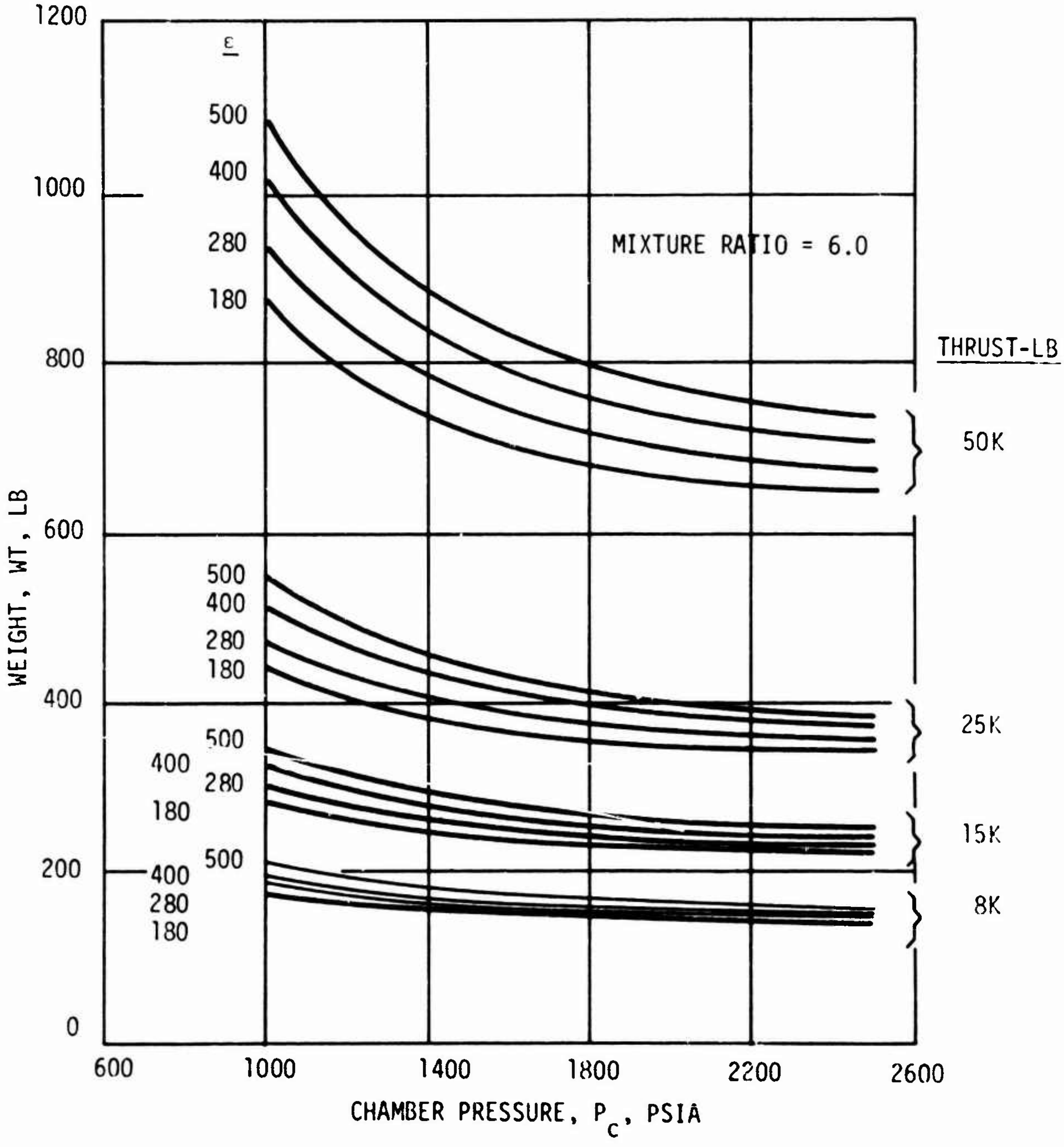


Figure 55. Engine Weight vs P_c, Fixed Nozzle, Staged Combustion Cycle

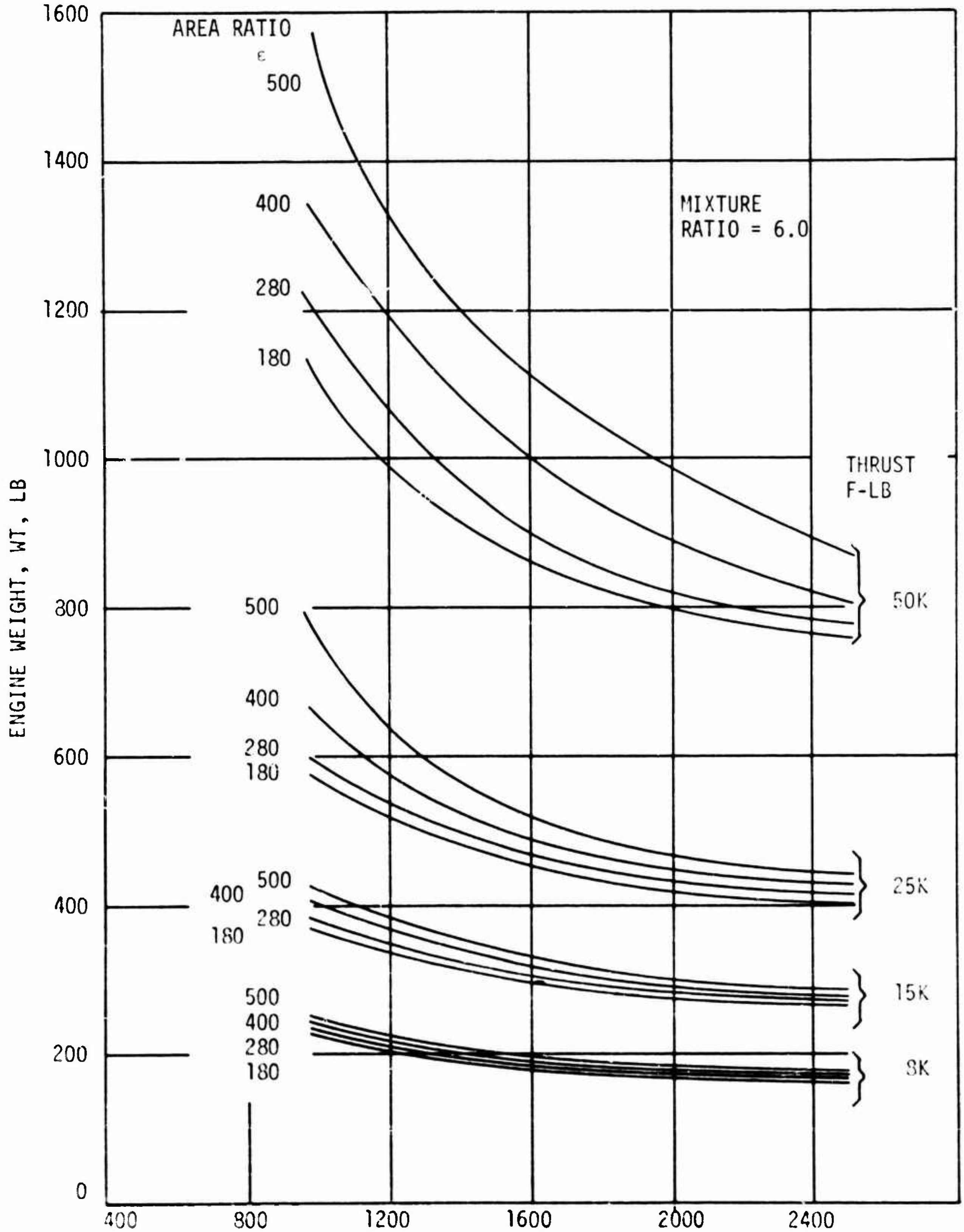


Figure 56. Engine Weight vs P_C, Minimum Weight Retractable Nozzle, Staged Combustion Cycle

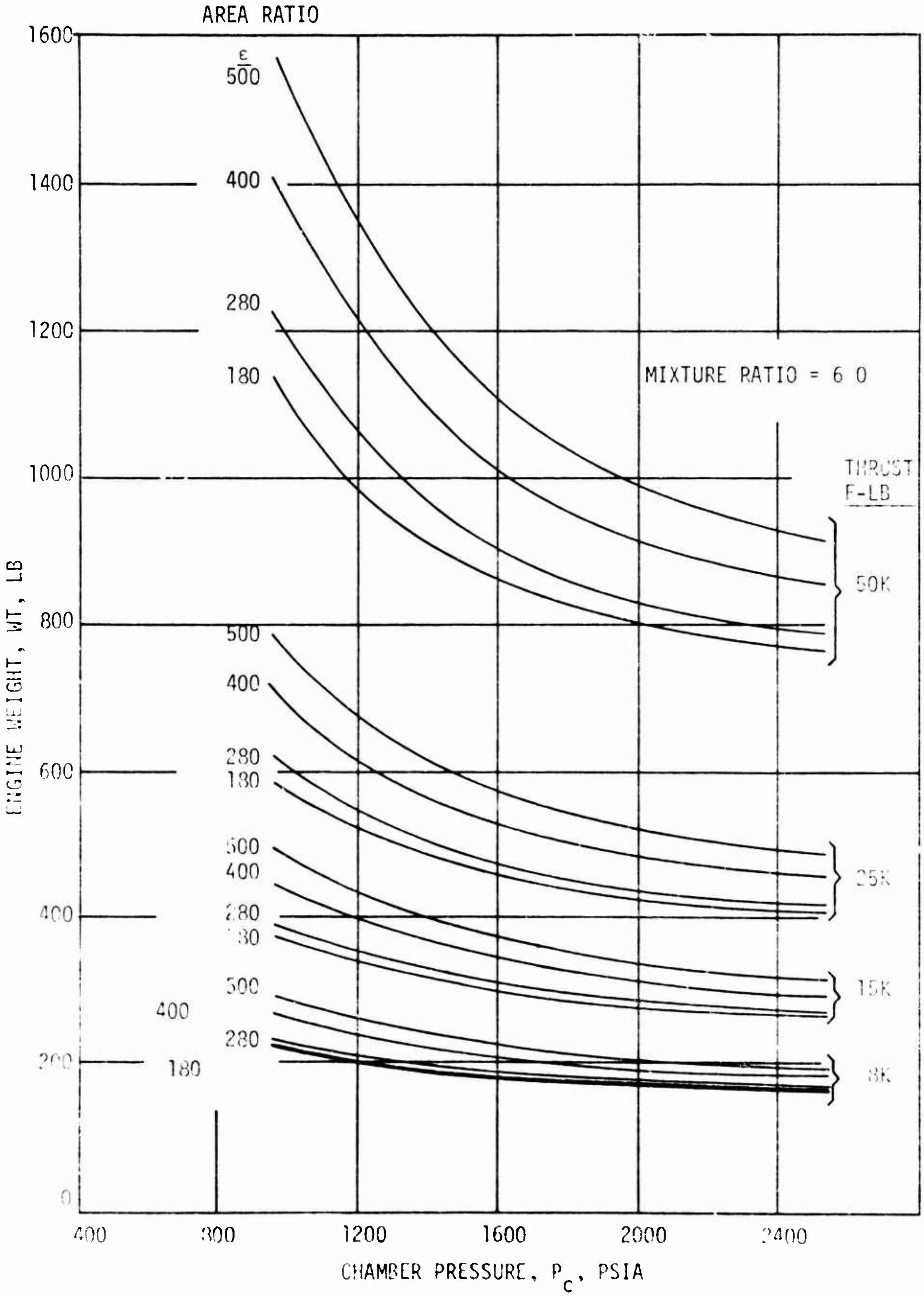


Figure 57. Engine Weight vs P_c, Minimum Length Retractable Nozzle, Staged Combustion Cycle

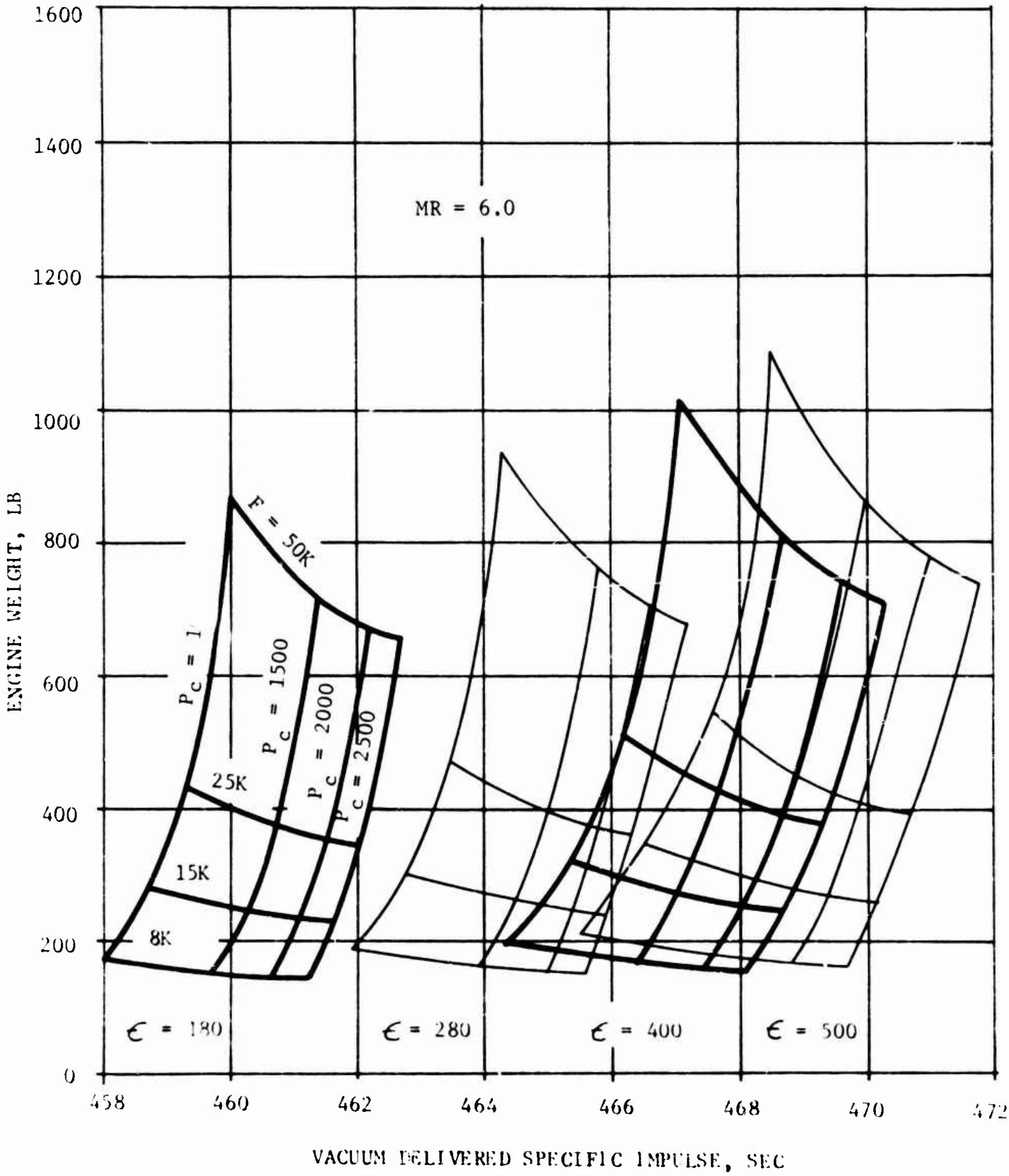


Figure 58. Engine Weight vs I_s, Fixed Nozzle, Staged Combustion Cycle

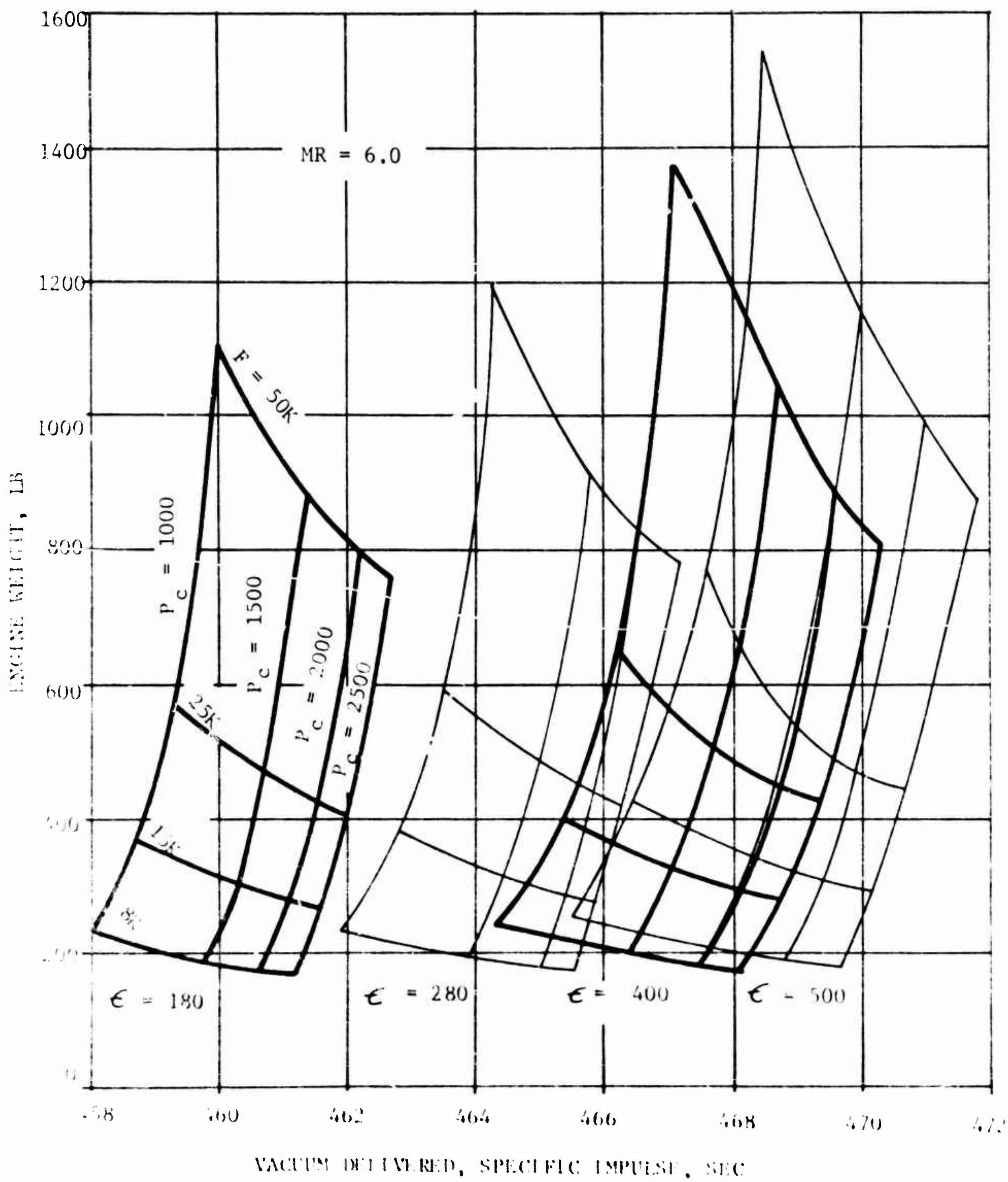


Figure 59. Engine Weight vs I_s, Minimum Weight Retractable Nozzle, Staged Combustion Cycle

(6) Stage Combustion Bleed Cycle

The configuration of the staged combustion bleed cycle is shown in Figure 60 and the schematic diagram in Figure 61. Engine performance, weight and interaction data is presented in Figures 62 through 70 and include low speed pump weights.

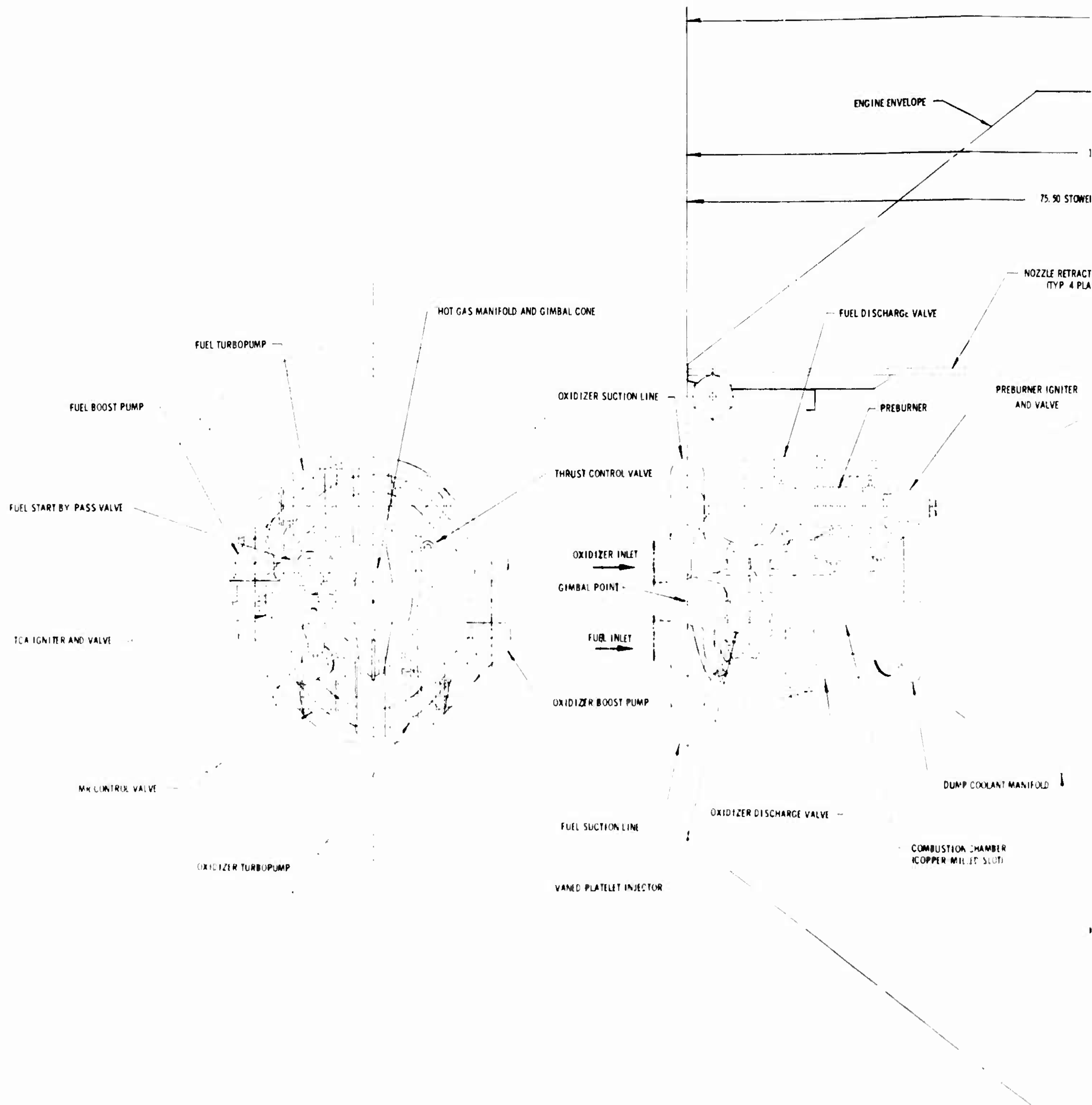
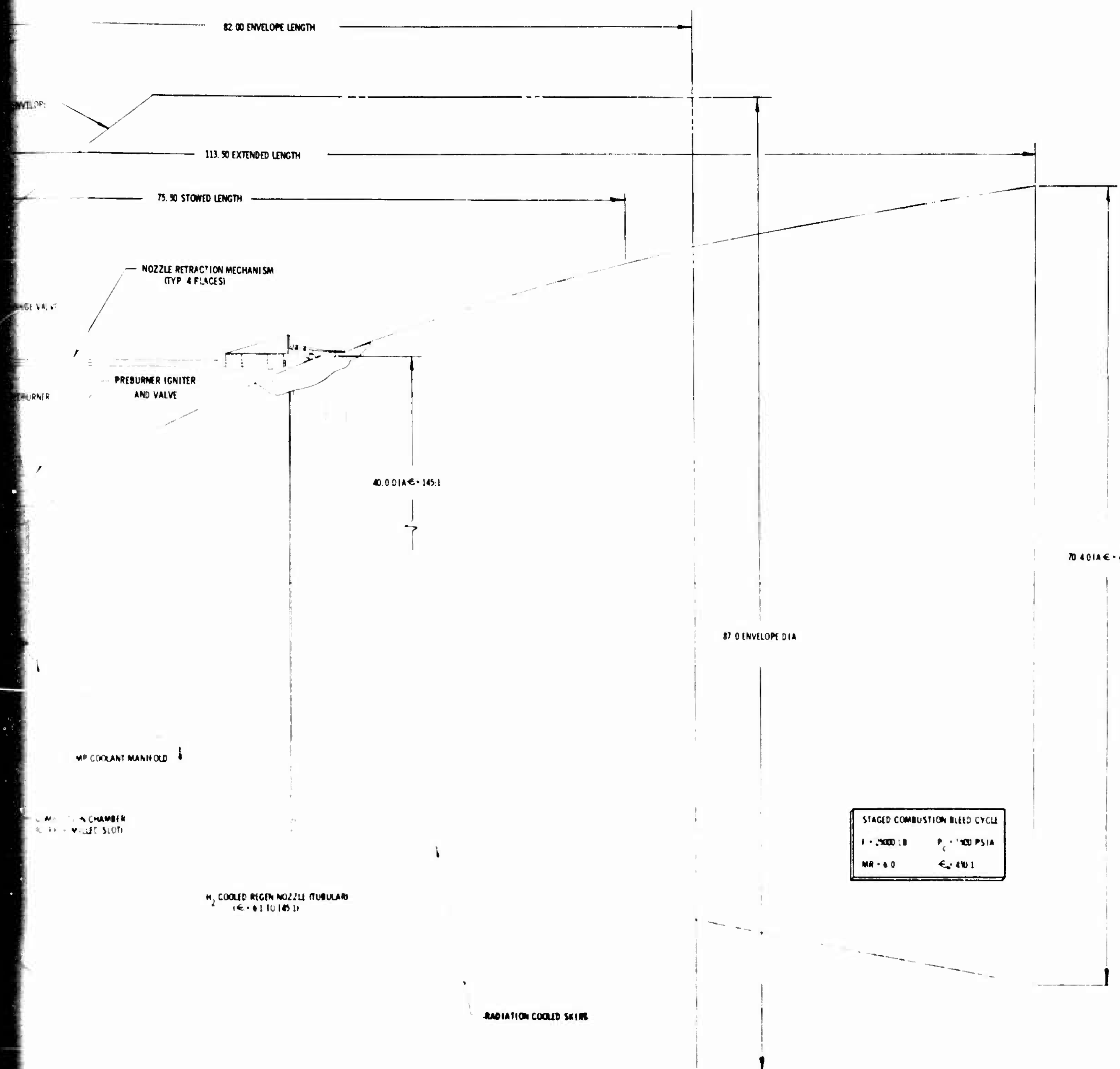


Figure 60. Configuration Drawing,



Drawing, Staged Combustion Bleed Cycle

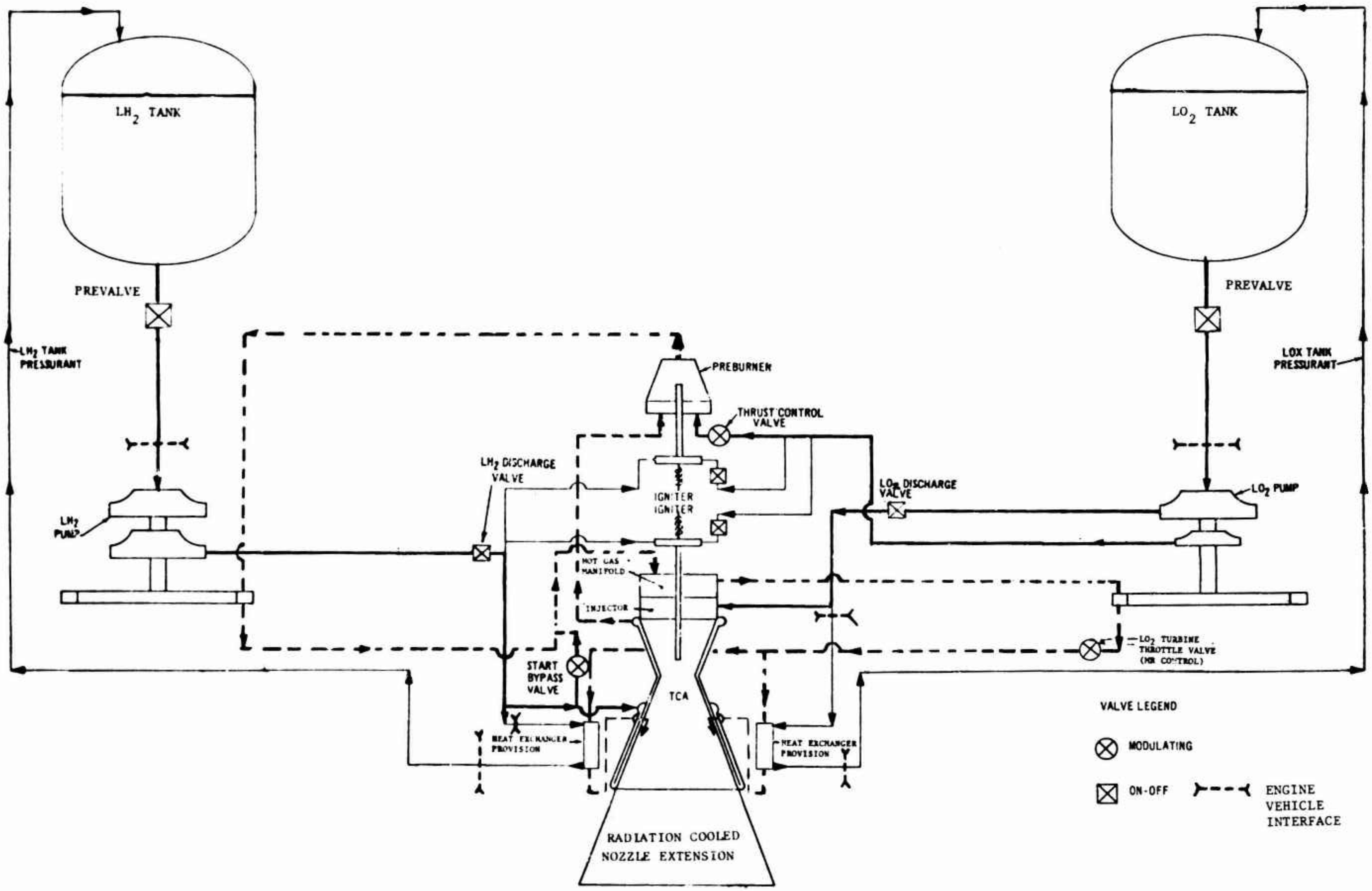


Figure 61. Engine Schematic, Staged Combustion Bleed Cycle

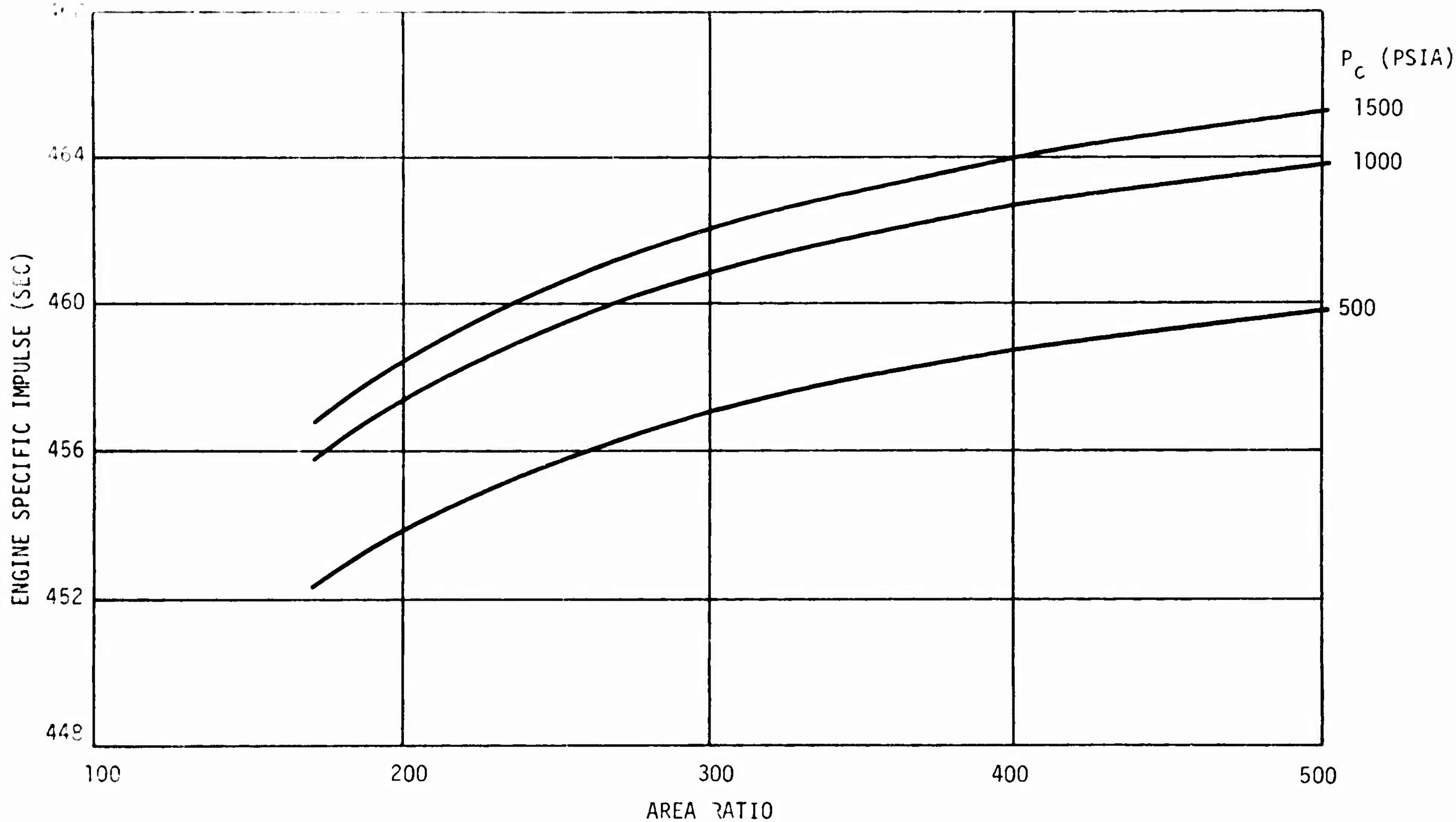


Figure 62. I_s vs P_c and Exit Area Ratio, MR = 6, F = 8K, Staged Combustion Bleed Cycle

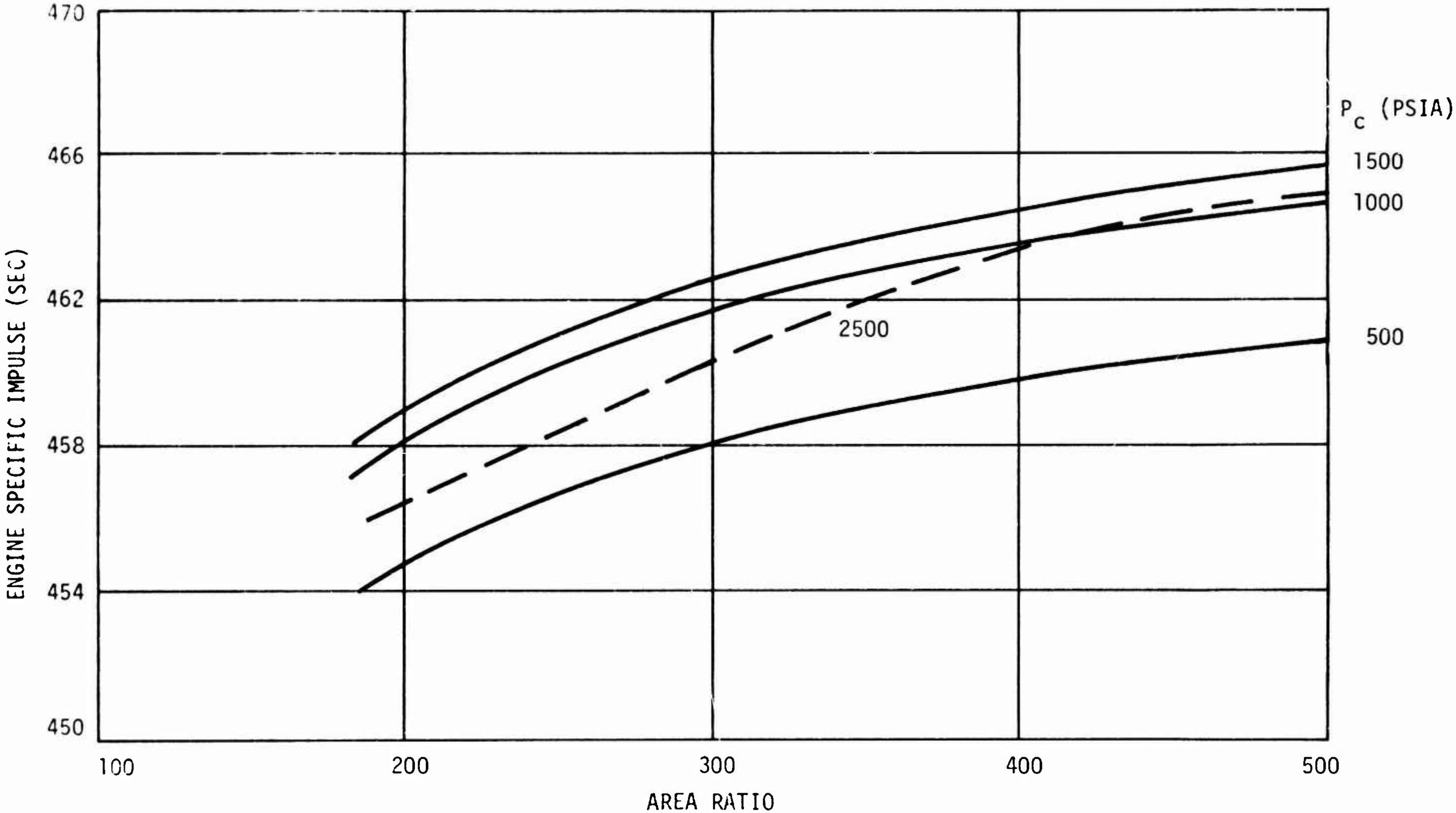


Figure 63. I_s vs P_c and Exit Area Ratio, MR = 6, F = 15K, Staged Combustion Bleed Cycle

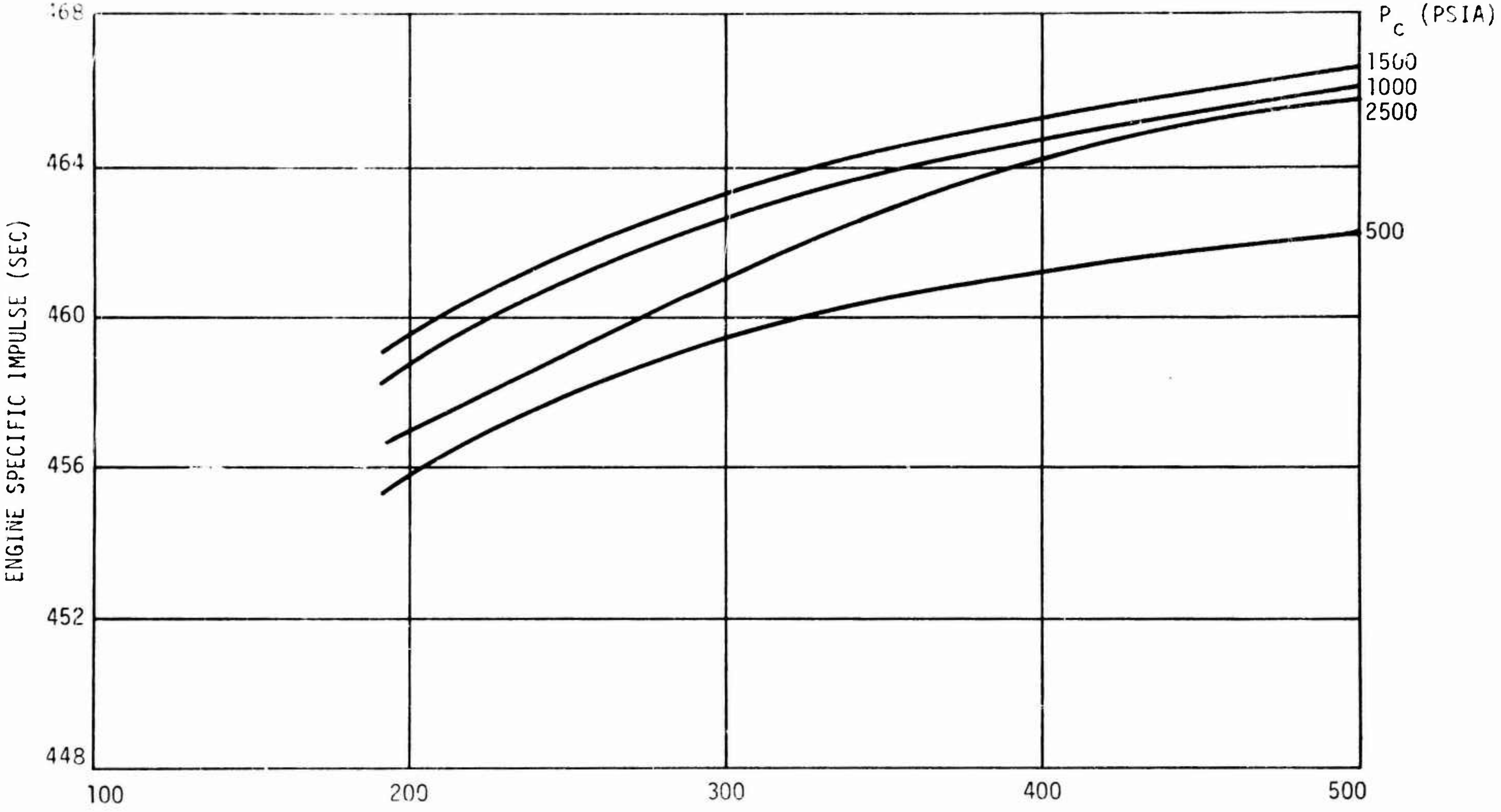


Figure 64. I_s vs P_c and Exit Area Ratio, MR = 6, F = 25K, Staged Combustion Bleed Cycle

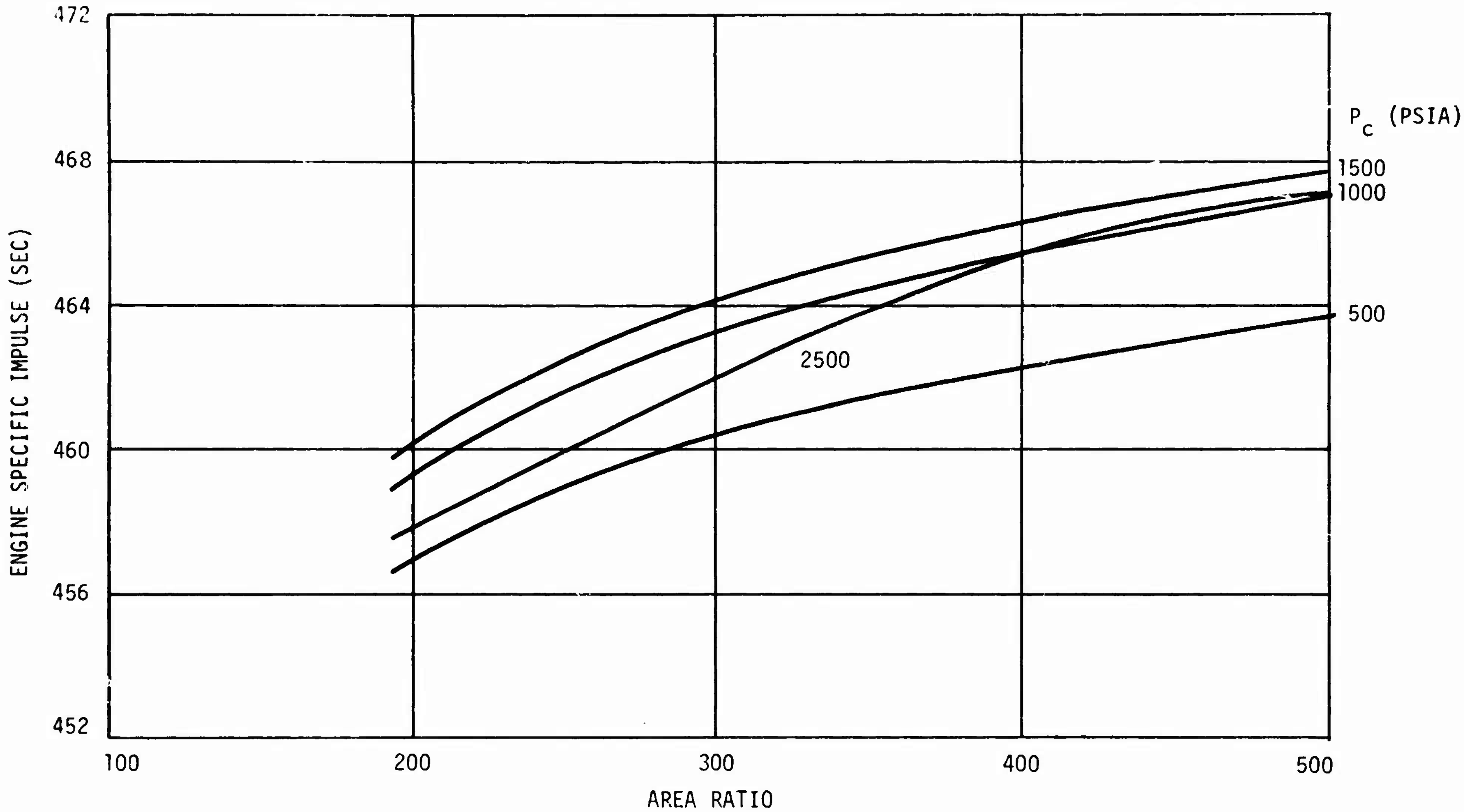
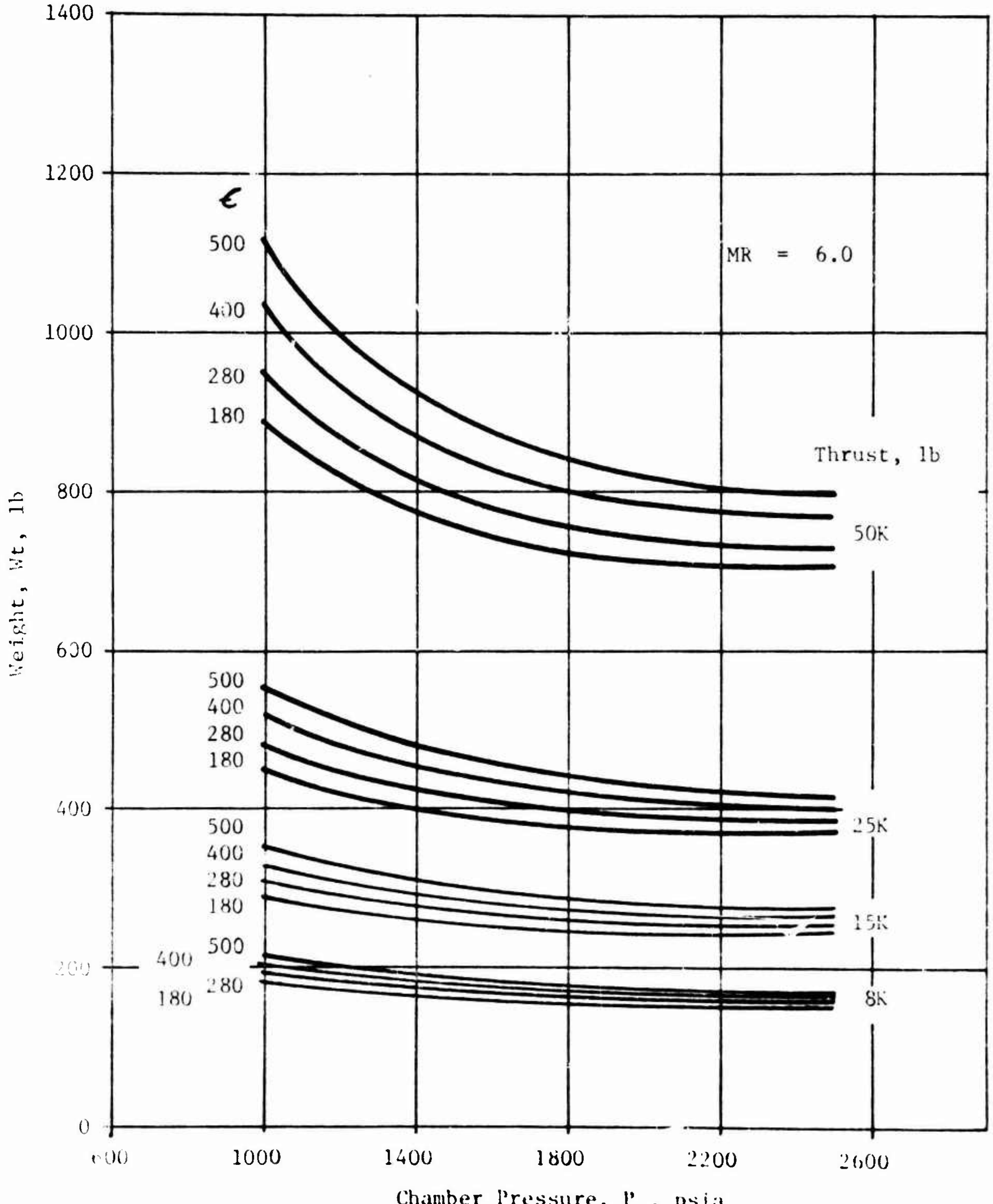


Figure 65. I_s vs P_c and Exit Area Ratio, MR = 6, F = 50K, Staged Combusti:n Bleed Cycle



Engine Weight vs P_c, Fixed Nozzle, Staged Combustion Bleed Cycle

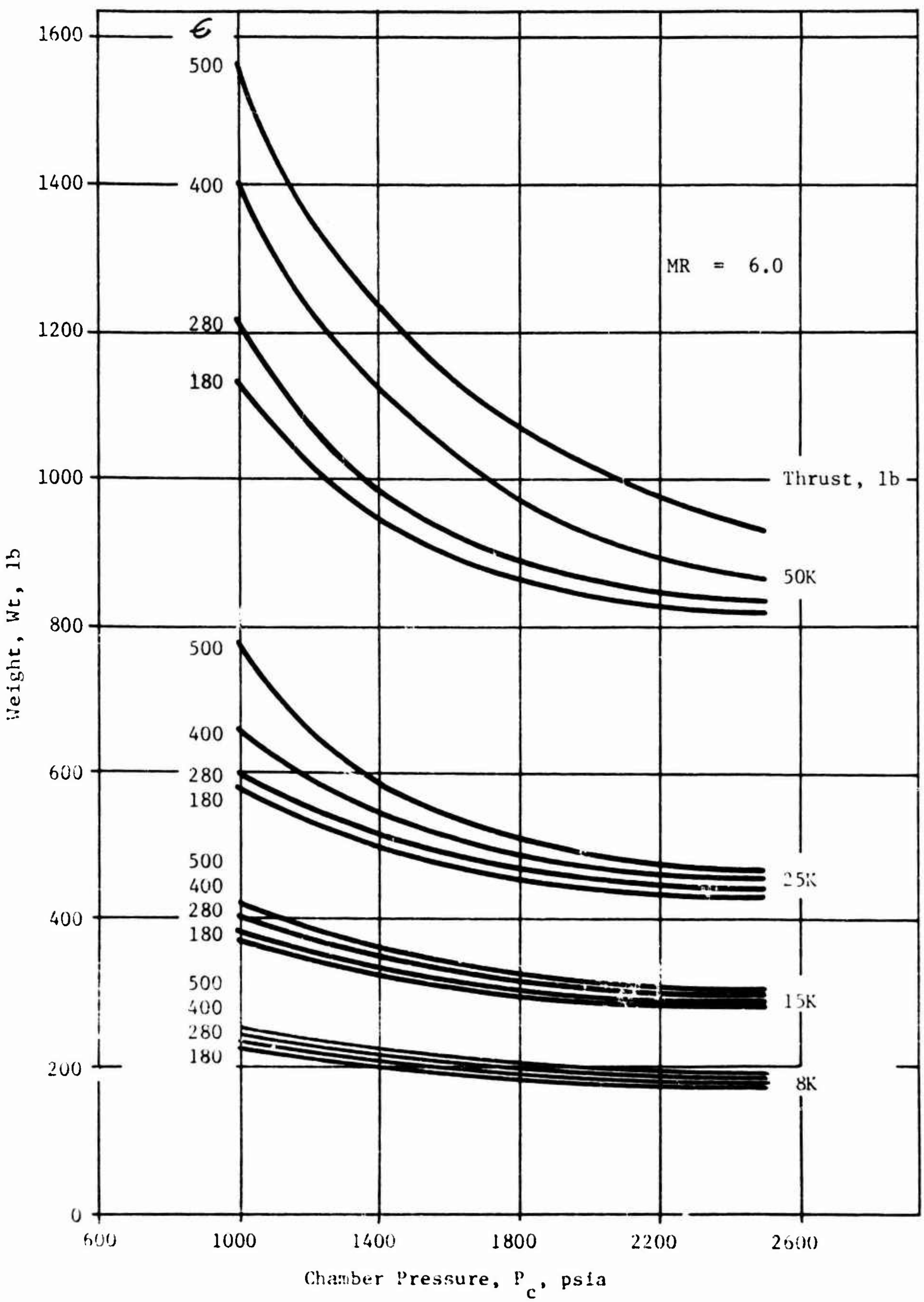


Figure 67. Engine Weight vs P_c, Minimum Weight Retractable Nozzle, Staged Combustion Bleed Cycle

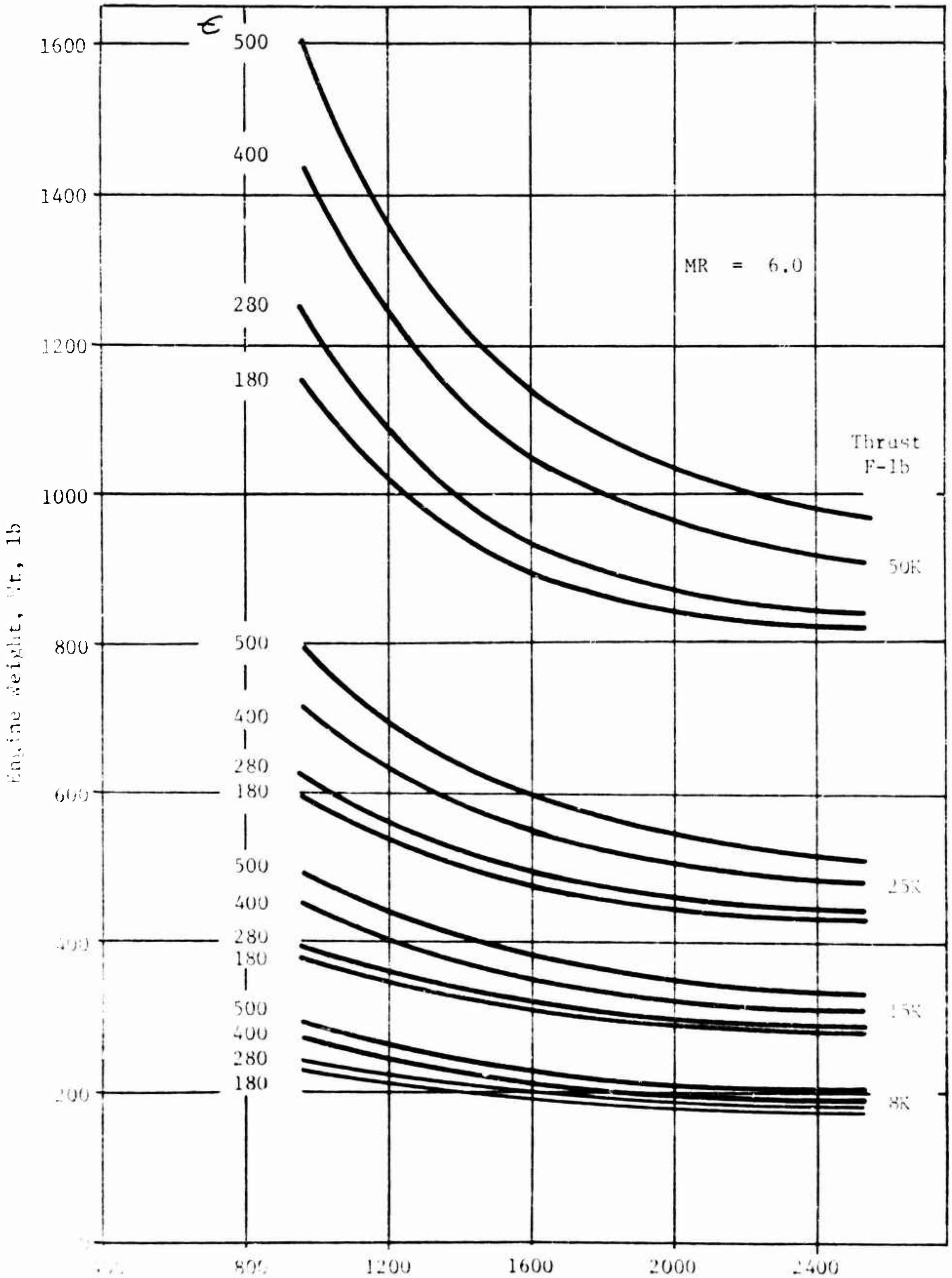


Figure 68. Engine Weight vs P_c, Minimum Length Retractable Nozzle, Staged Combustion Bleed Cycle

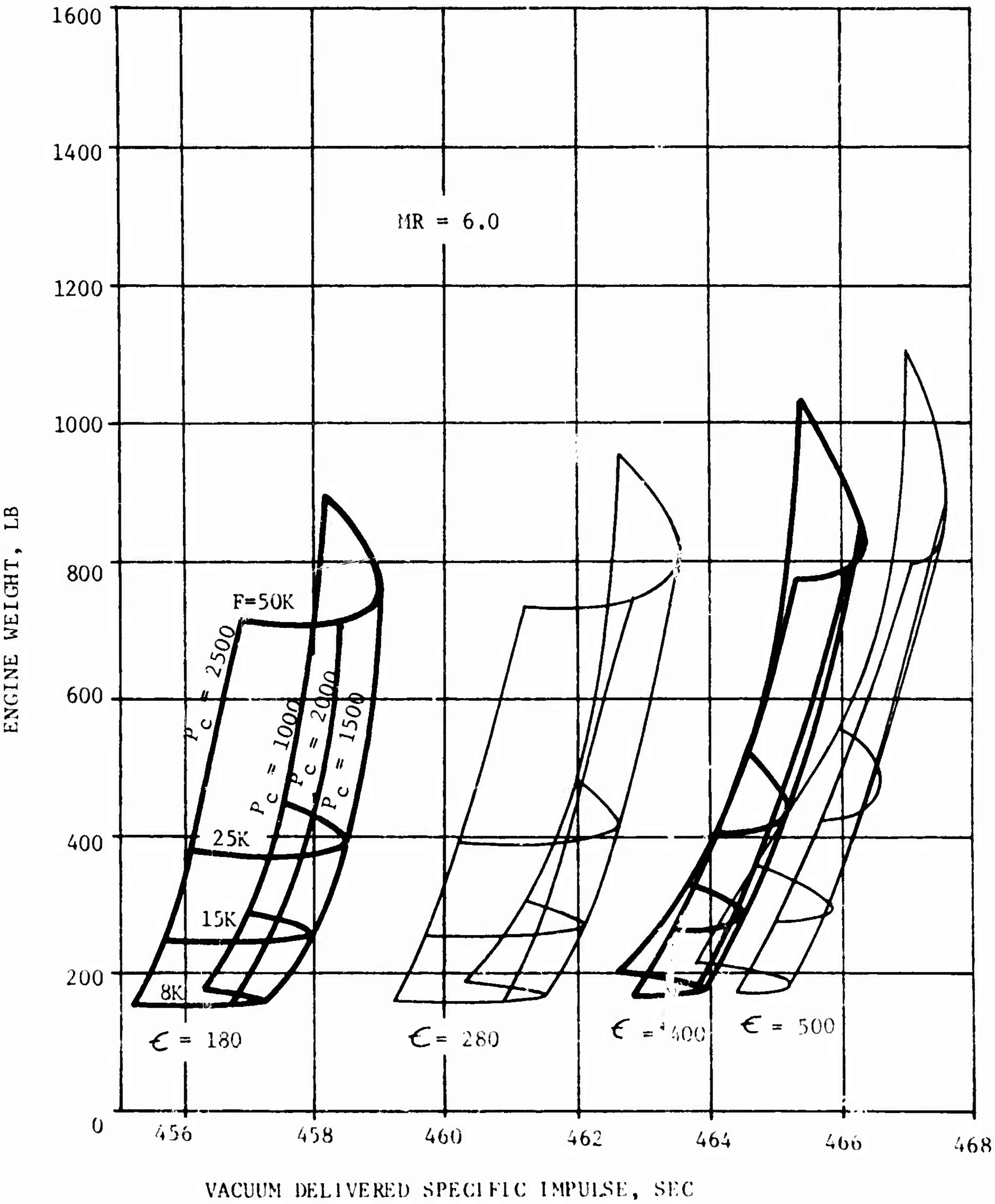


Figure 69. Engine Weight vs I_s, Fixed Nozzle, Staged Combustion Bleed Cycle

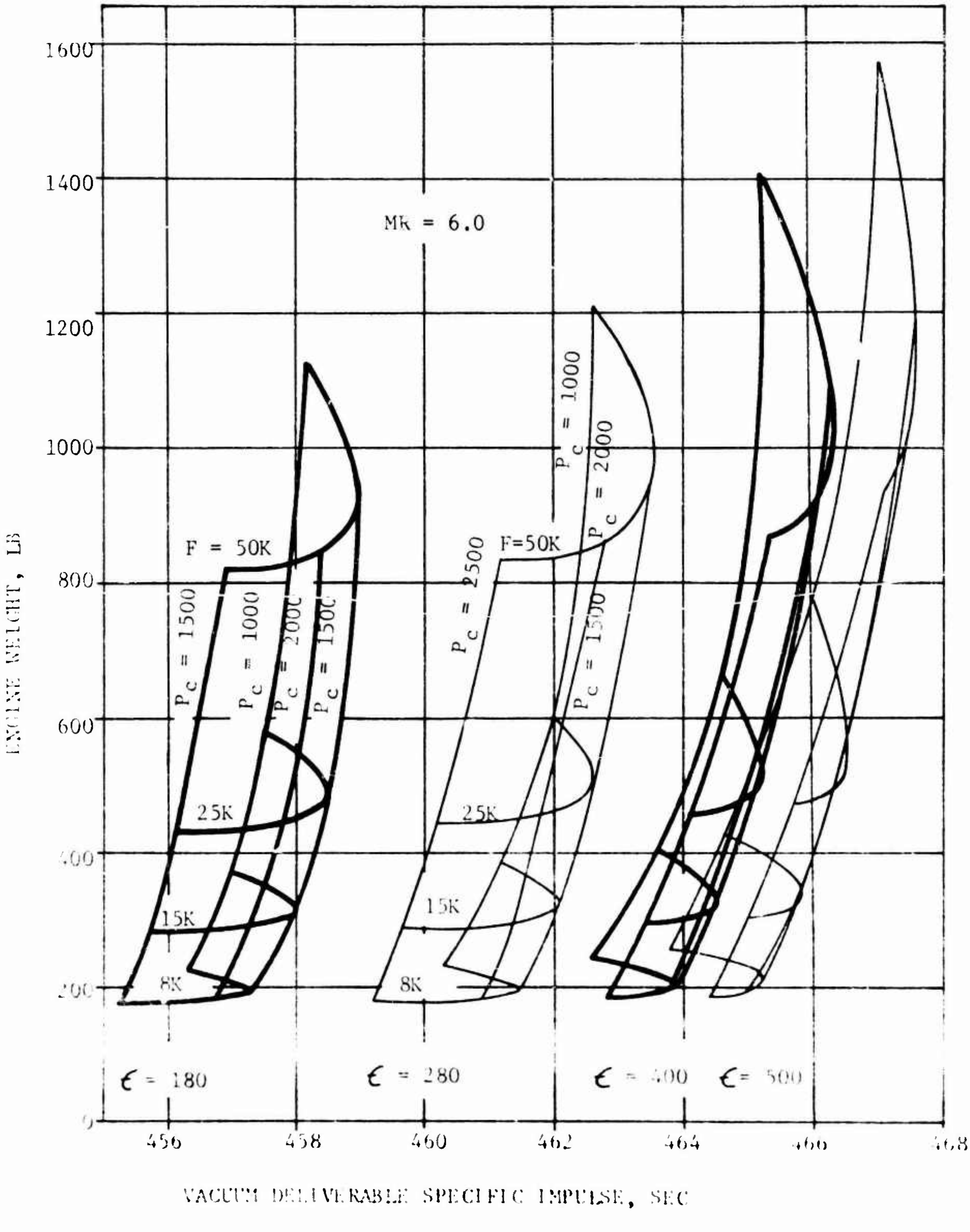


Figure [illegible]. Engine Weight vs I_s, Minimum Weight Retractable Nozzle, Staged Combustion Bleed Cycle

7. Engine Design Constraints

This section characterizes the engine cycles with respect to the engine constraints described in the Table of Requirements.

The main constraints are the engine life, envelope, and mission payload sensitivity parameters. All engines are designed to meet the throttle range of 5:1 and can operate over a mixture ratio of 5.5 to 6.5 at the required NPSH of 60 ft for the hydrogen pump and 16 ft for the oxygen pump.

a. Engine Throttling Range

Engine throttling is achieved by the control system selected for each cycle as presented in the appropriate flow schematic. Stability consideration consists of the selection of adequate injector stiffness for both the main and preburner or gas generator injectors.

b. Mixture Ratio Range

The mixture ratio range is achieved by the mixture ratio control valve and the selection of adequate coolant pressure drop to facilitate adequate cooling over the whole operating range. Injector stiffness is sufficient to achieve stable operation over the required mixture ratio range at any thrust level within the required throttling range.

c. Engine Life Requirements

In the parametric analysis, engine life is constrained by the life of the thrust chamber since it strongly influences the maximum operating chamber pressure at each thrust level. To obtain the maximum payload capability of each cycle within the envelope limitation, the chamber pressure must be selected as high as possible. High pressure permits high expansion nozzle area ratios with minimum engine weight. The absolute potential level of chamber pressure within the life limit is strongly influenced by the chamber material selected.

Zirconium-Copper (ZrCu) was selected for all engine cycles. This selection is based on sample tests conducted on an Aerojet material investigation program. The low cycle fatigue capability of ZrCu is summarized in Figure 9, indicating the low thermal fatigue life cycle as a function of the gradient across the thrust chamber wall in the throat plane.

Based on these data, it was assumed that the temperature gradient at the wall must be restricted to $\Delta Tw = 935°F$. This assumption results in a safety factor SF = 4.0 at the upper data limit and a SF = 2.0 at the lower data limit. The flame side wall temperature Tw is limited to $Tw = 1000°F$.

Since the low cycle fatigue life shown in Figure 9 is probably the most important assumption within this parametric analysis, a conservative approach has been taken. Unfortunately, the supporting data is limited and should be confirmed under actual operating conditions. Consequently, it is recommended for future technology investigation.

The chamber life is obtained by designing the thrust chamber for ΔTw = 935°F and Tw_{max} = 1000°F. This is achieved by selection of the proper coolant mach number at the throat location which in turn sets the chamber coolant pressure drop requirements.

The analysis of the coolant pressure drop requirements was the subject of a very involved heat transfer analysis which is presented in Appendix B, and summarized in Figures 71 and 72 as functions of thrust level, chamber pressure, and coolant outlet pressure.

d. Engine Envelope

The engine envelope defines the stowed engine envelope. This requirement restricts the fixed engine length to an overall length of 82 inches. The achievable nozzle expansion area ratios are dependent upon the nozzle contour and thrust chamber length. For the retractable nozzles, the achievable expansion area ratios are also dependent on the retraction (i.e., minimum weight or minimum length).

The achievable expansion area ratios were computed for all three nozzle types as function of thrust/chamber pressure ratio, and various chamber lengths and are presented in Figures 73 through 80 for MR = 6.0.

In addition, for the retractable nozzles, the area ratios at which the division between the fixed and retractable section is located are shown in Figure 75 through Figure 77 for the minimum length nozzle and Figures 78 through 80 for the minimum weight retractable nozzle.

e. Power Balance for Topping Cycles

A power balance analysis for the topping cycles, including the expander and staged combustion cycles, was conducted using the coolant pressure drops required to meet the low cycle fatigue life of the thrust chamber. The results of this analysis are shown in Figures 81 and 82, yielding fuel pump discharge requirements as function of thrust level and chamber pressures. This analysis was used to assign the proper chamber pressures to achieve adequate power balance capability. 80% of the maximum achievable chamber pressure was selected for the staged combustion cycle and 90% for the expander cycle.

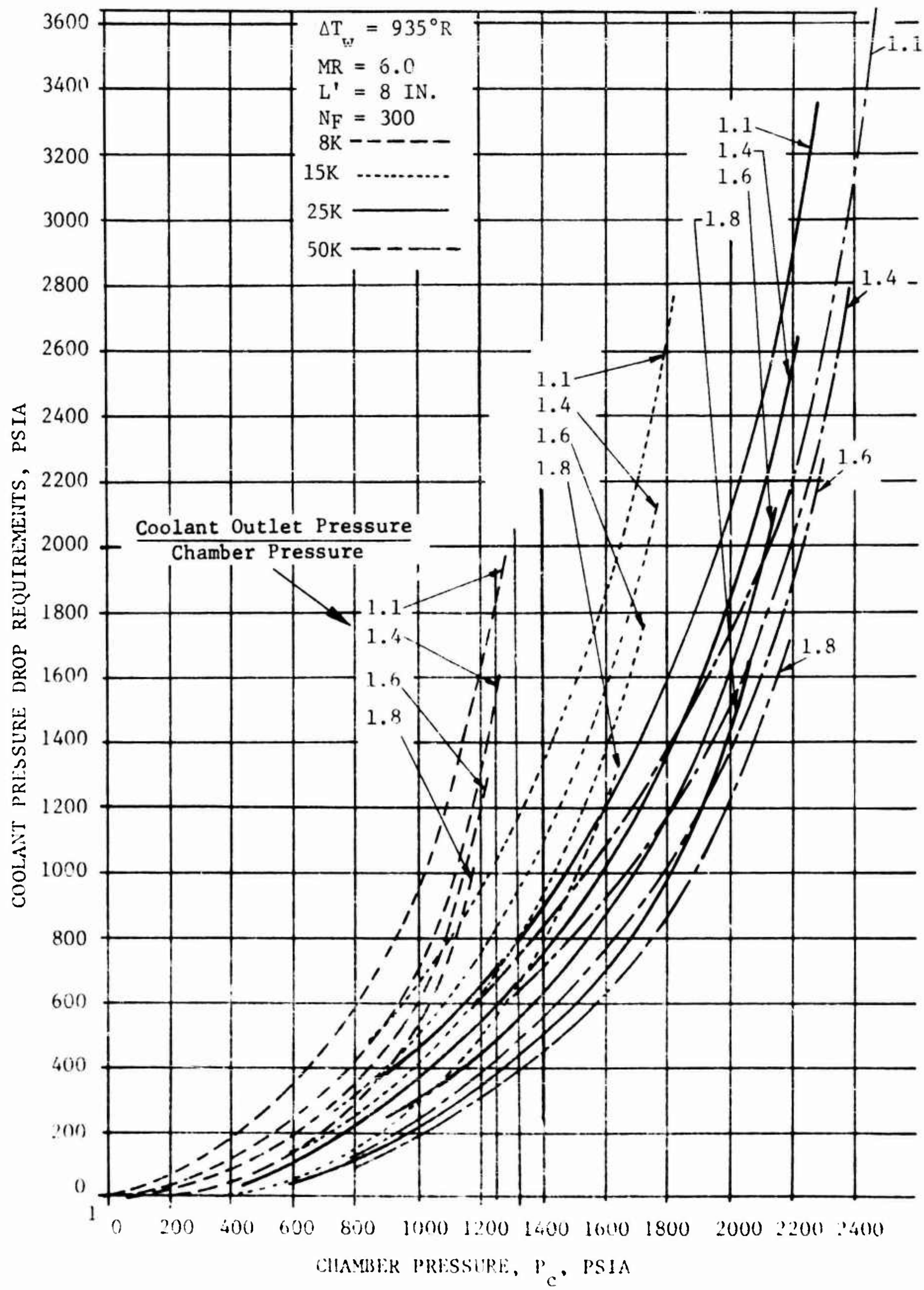


Figure 71. Coolant Pressure Drop vs P_c

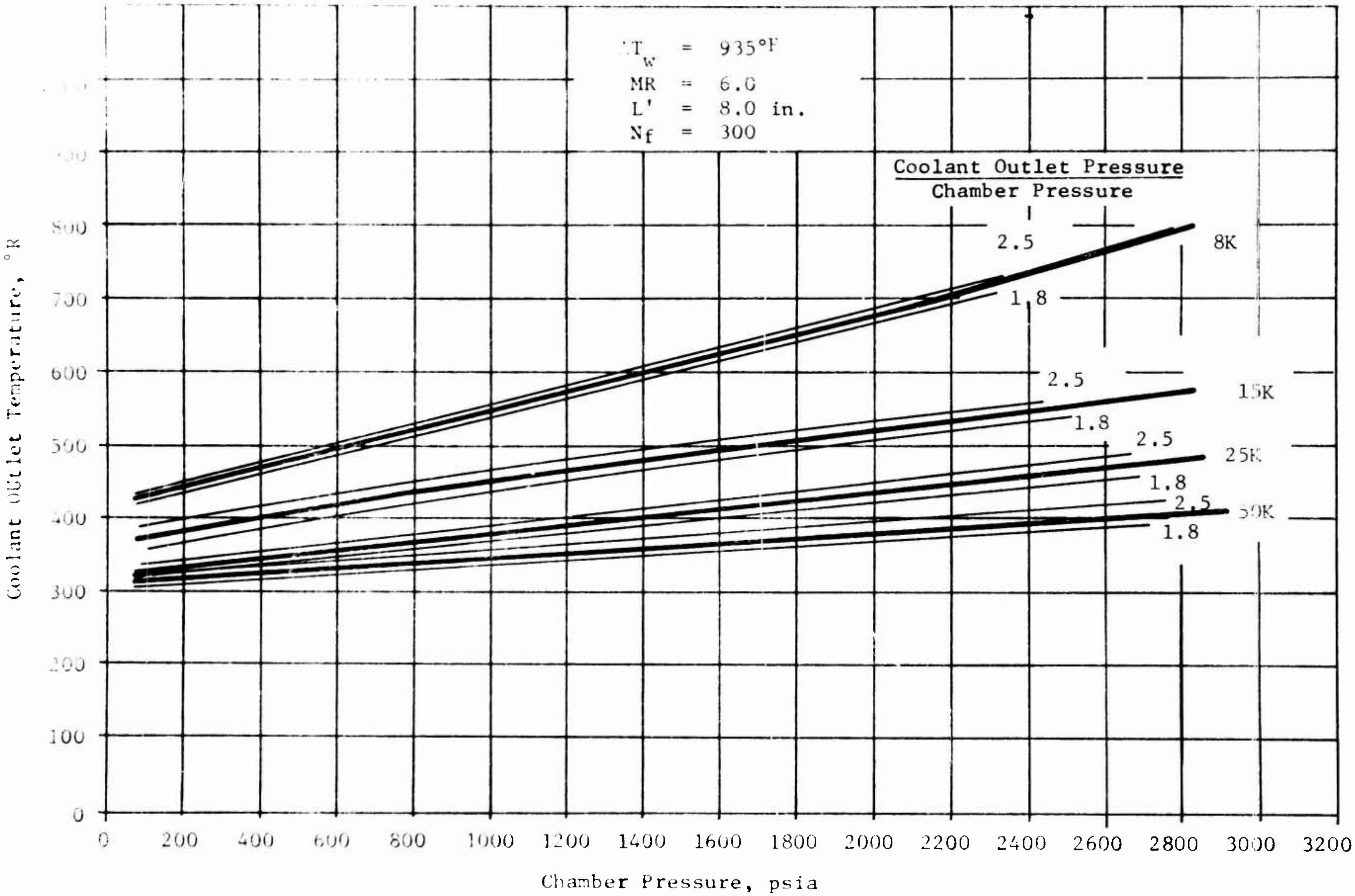


Figure 72. Coolant Temperature Rise vs P_c

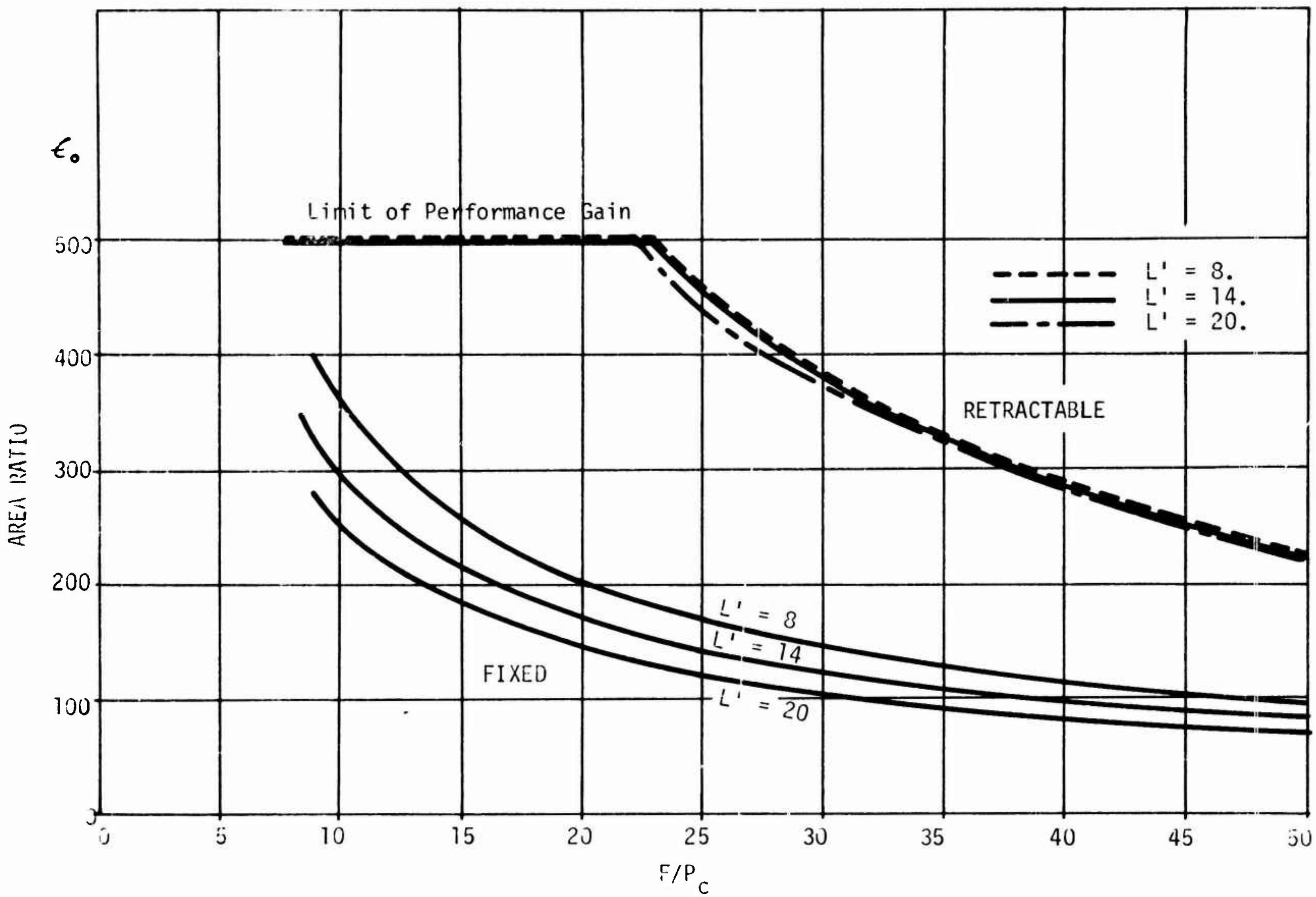


Figure 73. Maximum Area Ratio for Fixed Nozzles and Minimum Stowed Length Retractable Nozzles

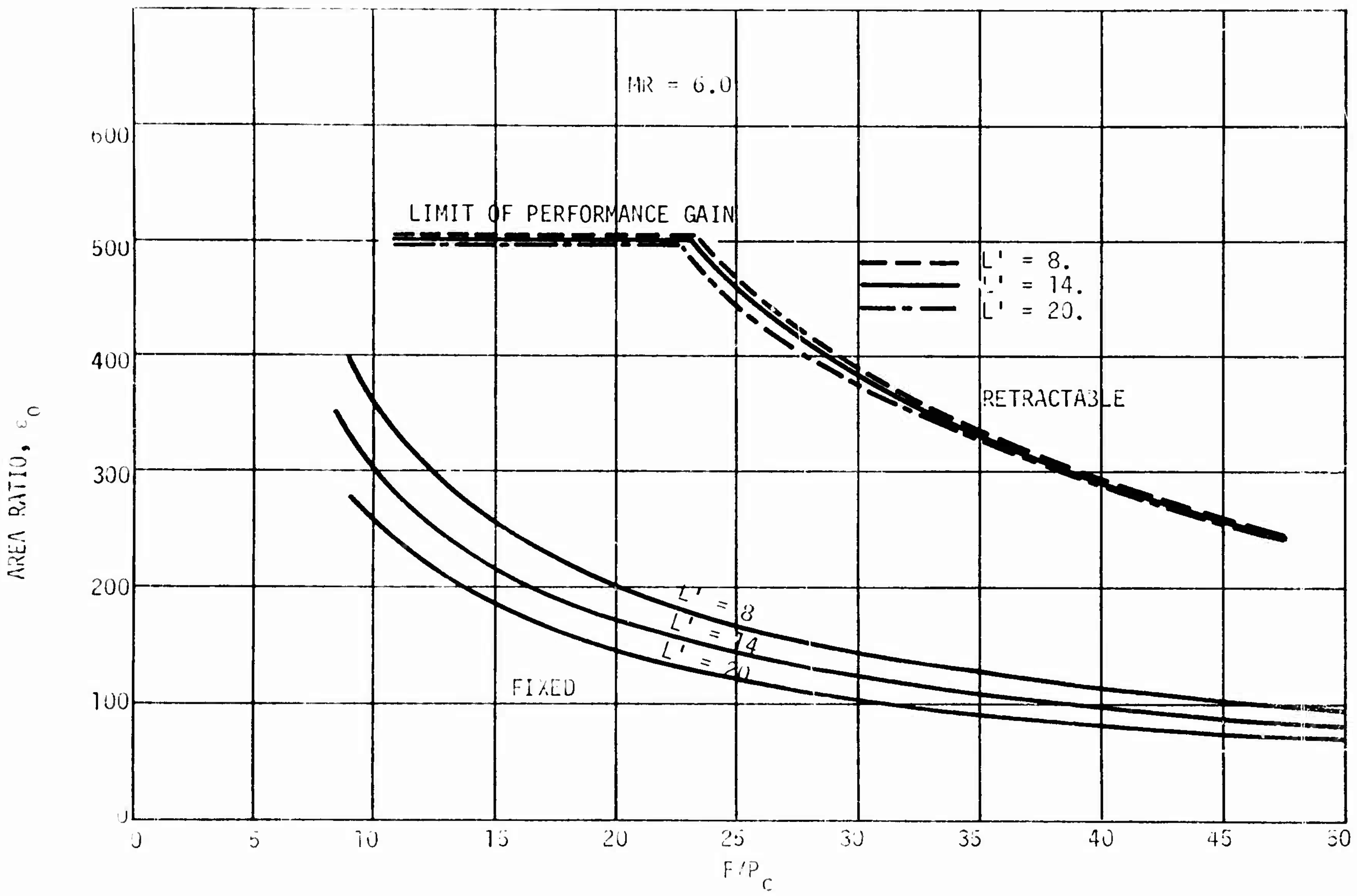


Figure 74. Maximum Area Ratio for Fixed Nozzles and Minimum Weight Retractable Nozzles

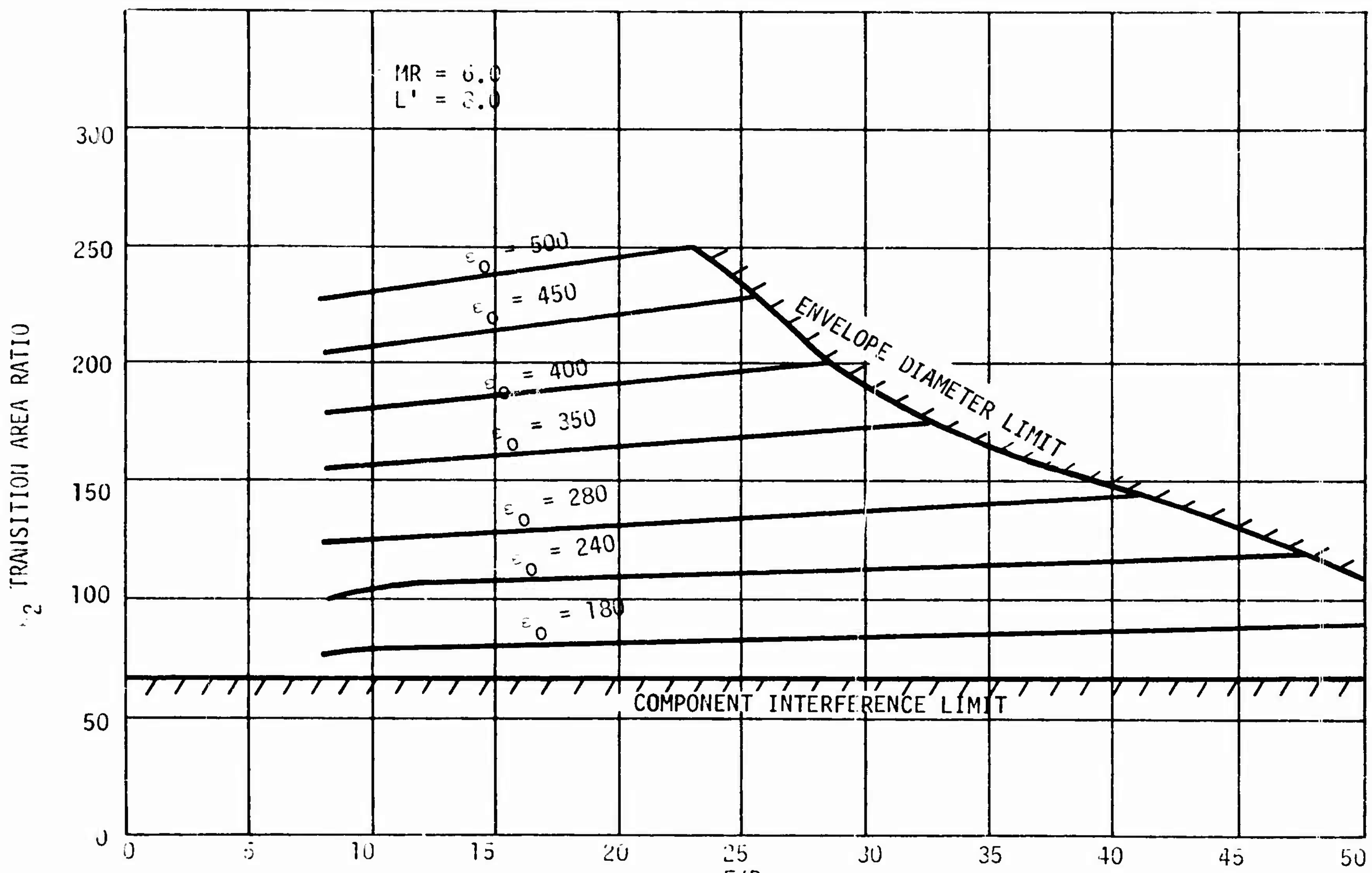


Figure 75. Transition Area Ratios for Minimum Stowed Length Retractable Nozzle, L' = 8-in.

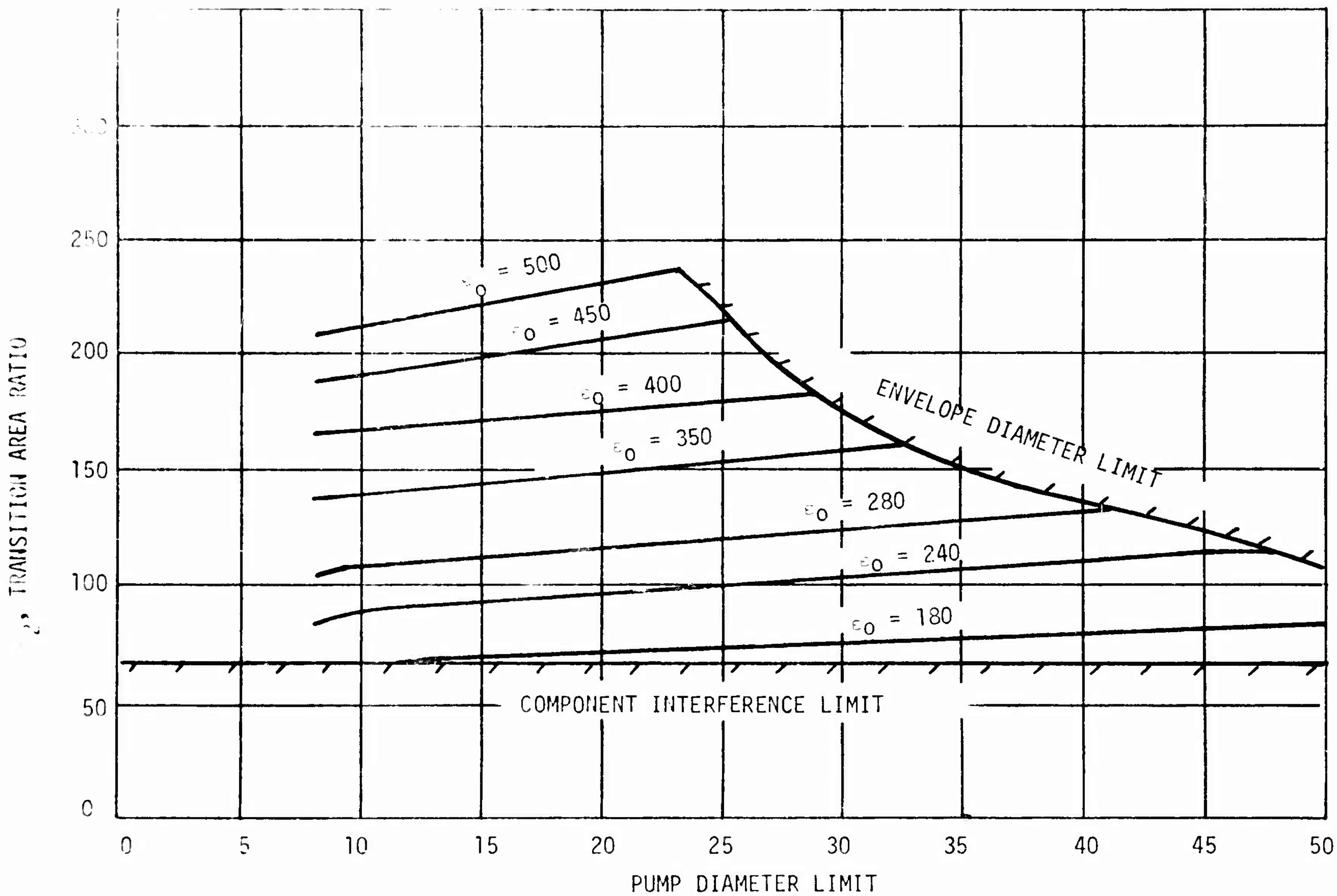


Figure 76. Transition Area Ratios for Minimum Stowed Length Nozzle, L' = 14-in.

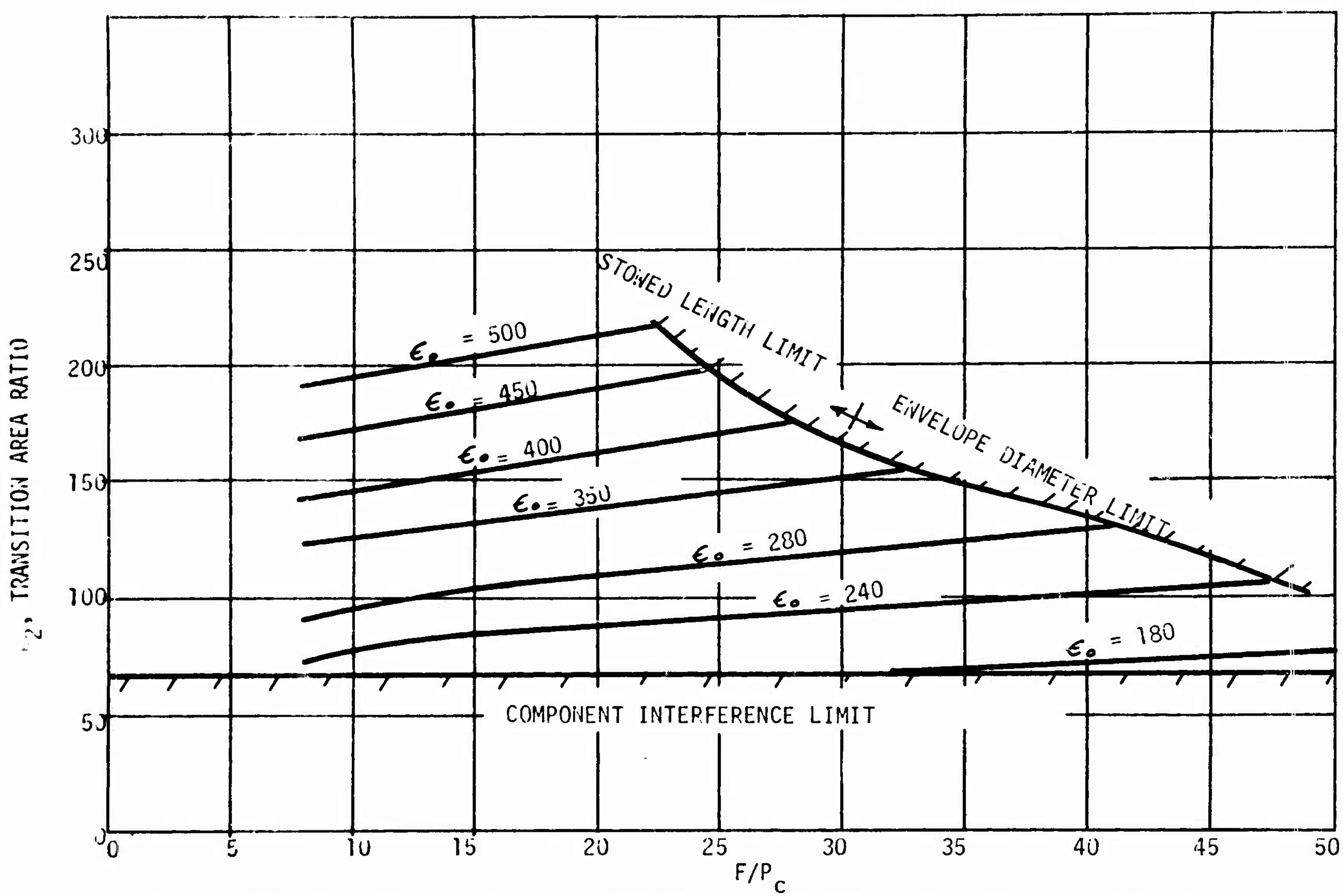


Figure 77. Transition Area Ratios for Minimum Stowed Length Nozzle, L' = 20-in.

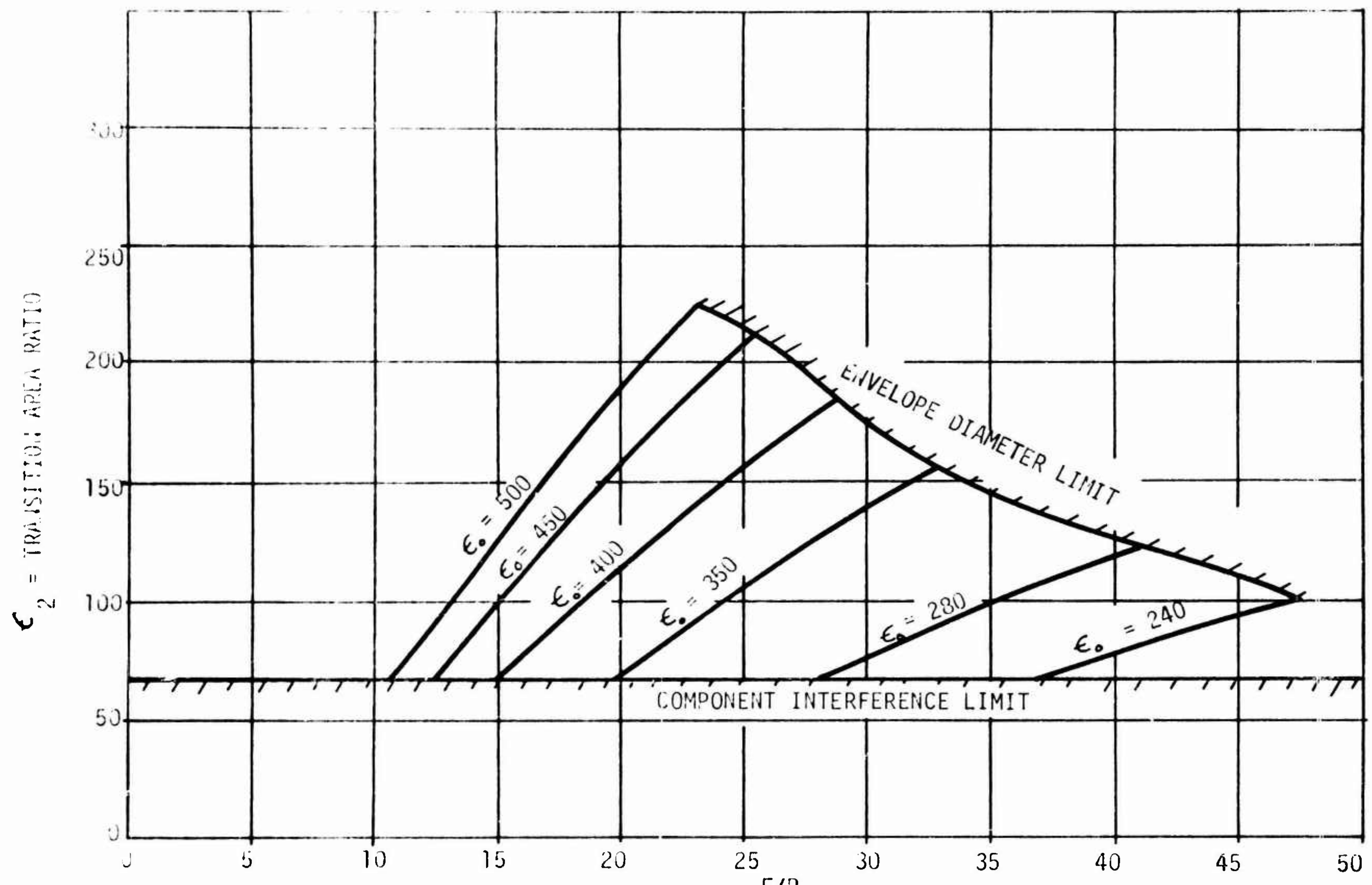


Figure 78. Transition Area Ratios for Minimum Weight Retractable Nozzle, L' = 8-in.

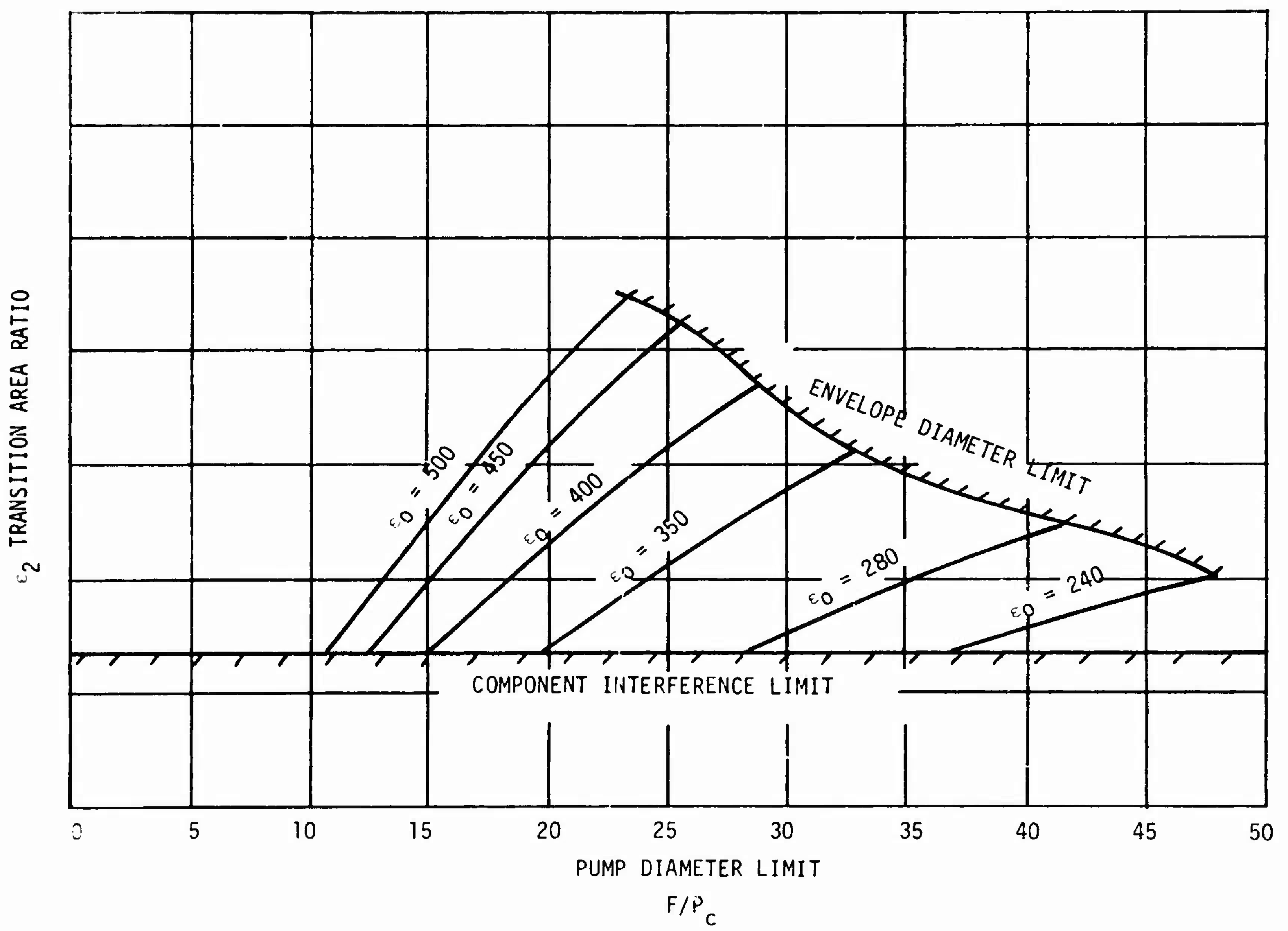


Figure 79. Transition Area Ratios for Minimum Weight Retractable Nozzle, L' = 14-in.

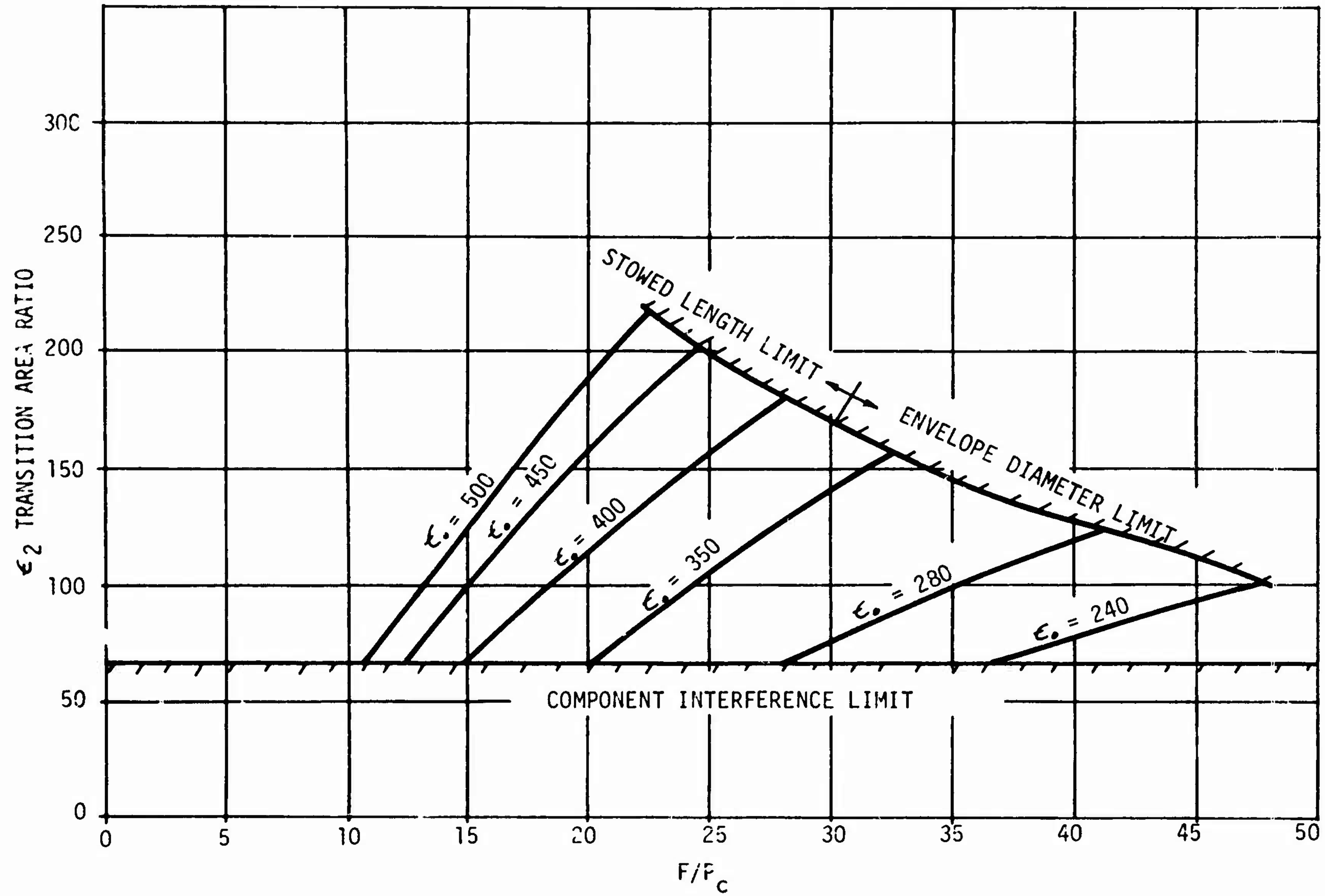


Figure 80. Transition Area Ratios for Minimum Weight Retractable Nozzle, L' = 20-in.

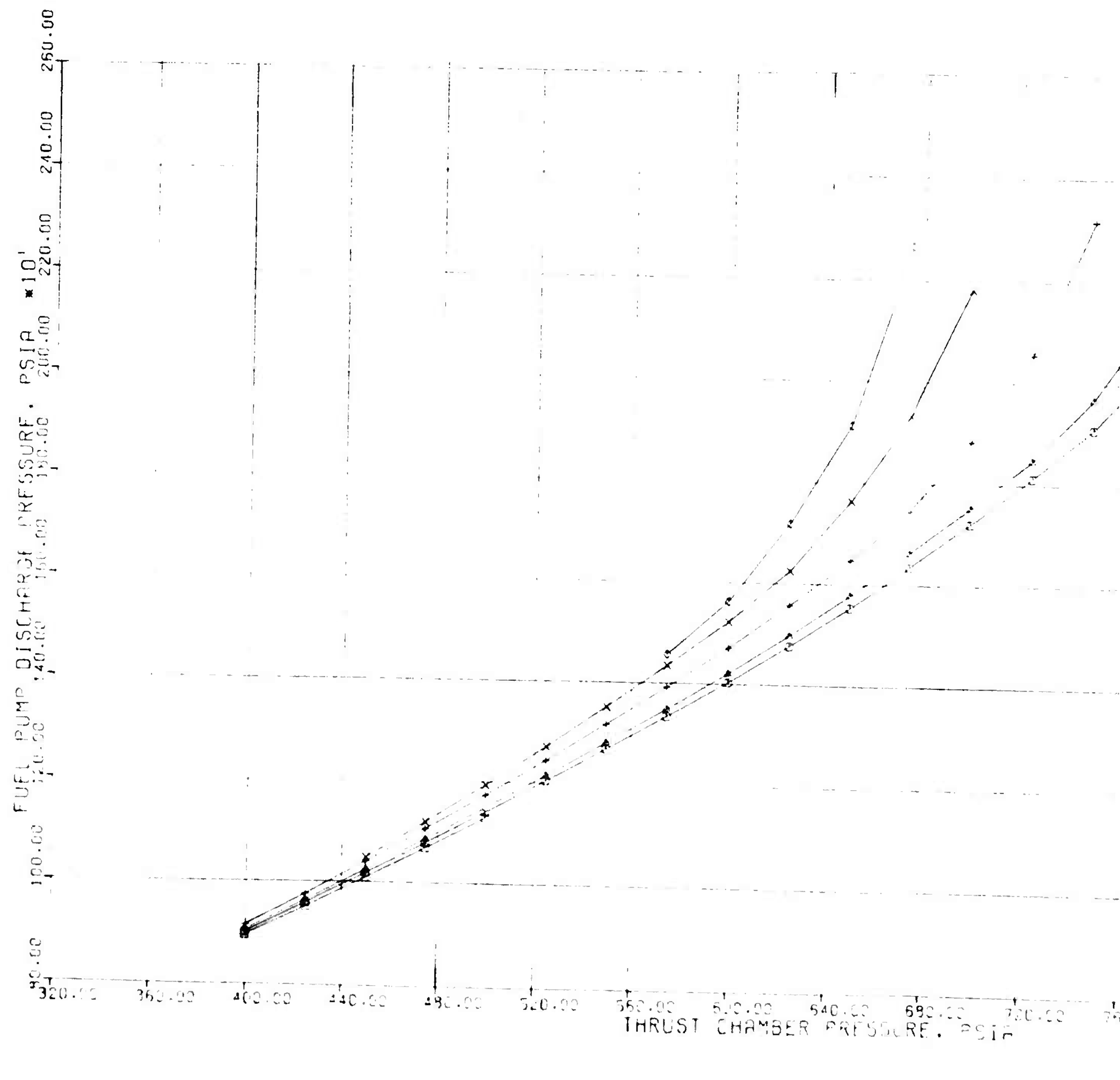
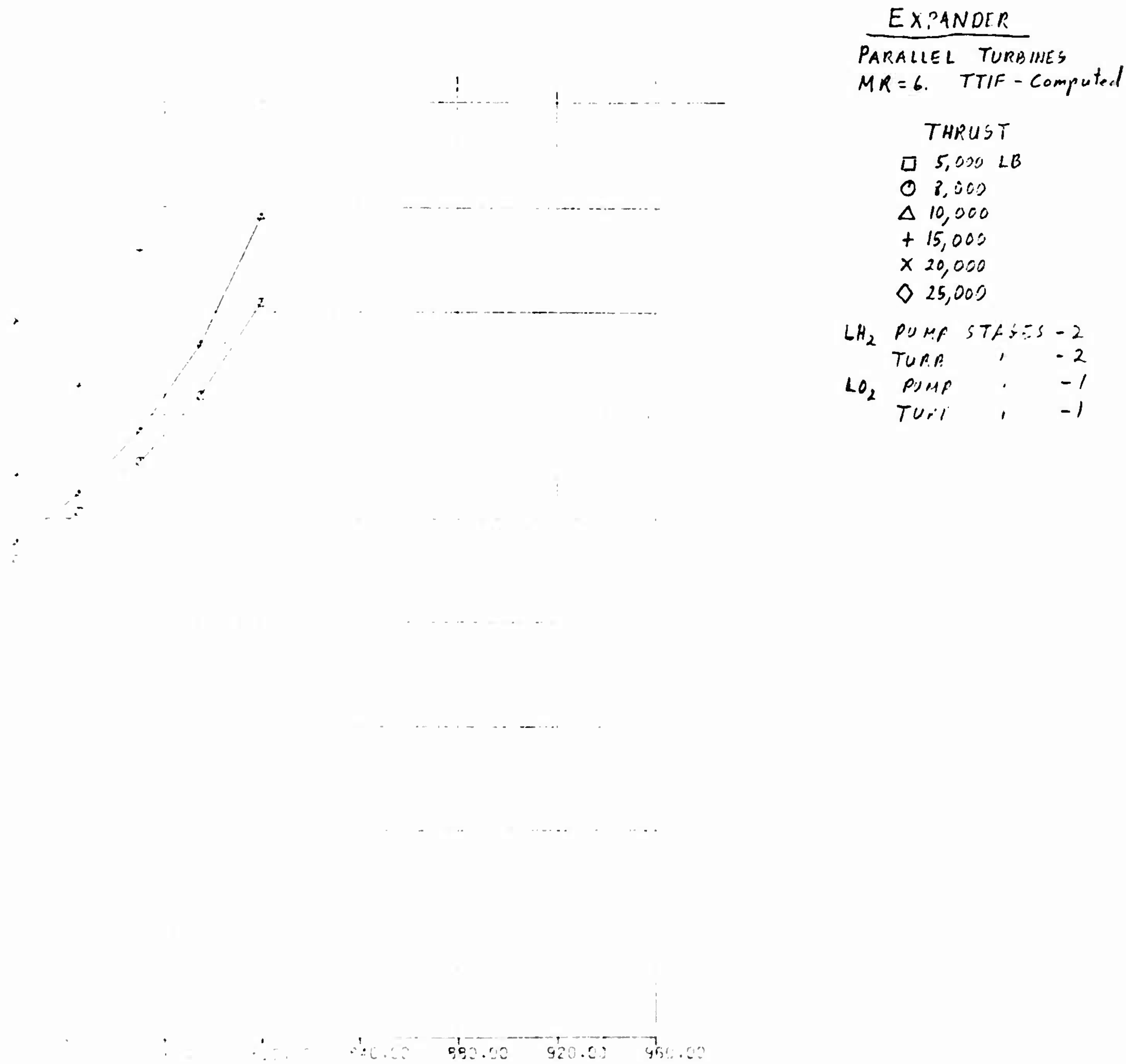


Figure 81. Power Balance for Expander C



...xpander Cycle, MR = 6

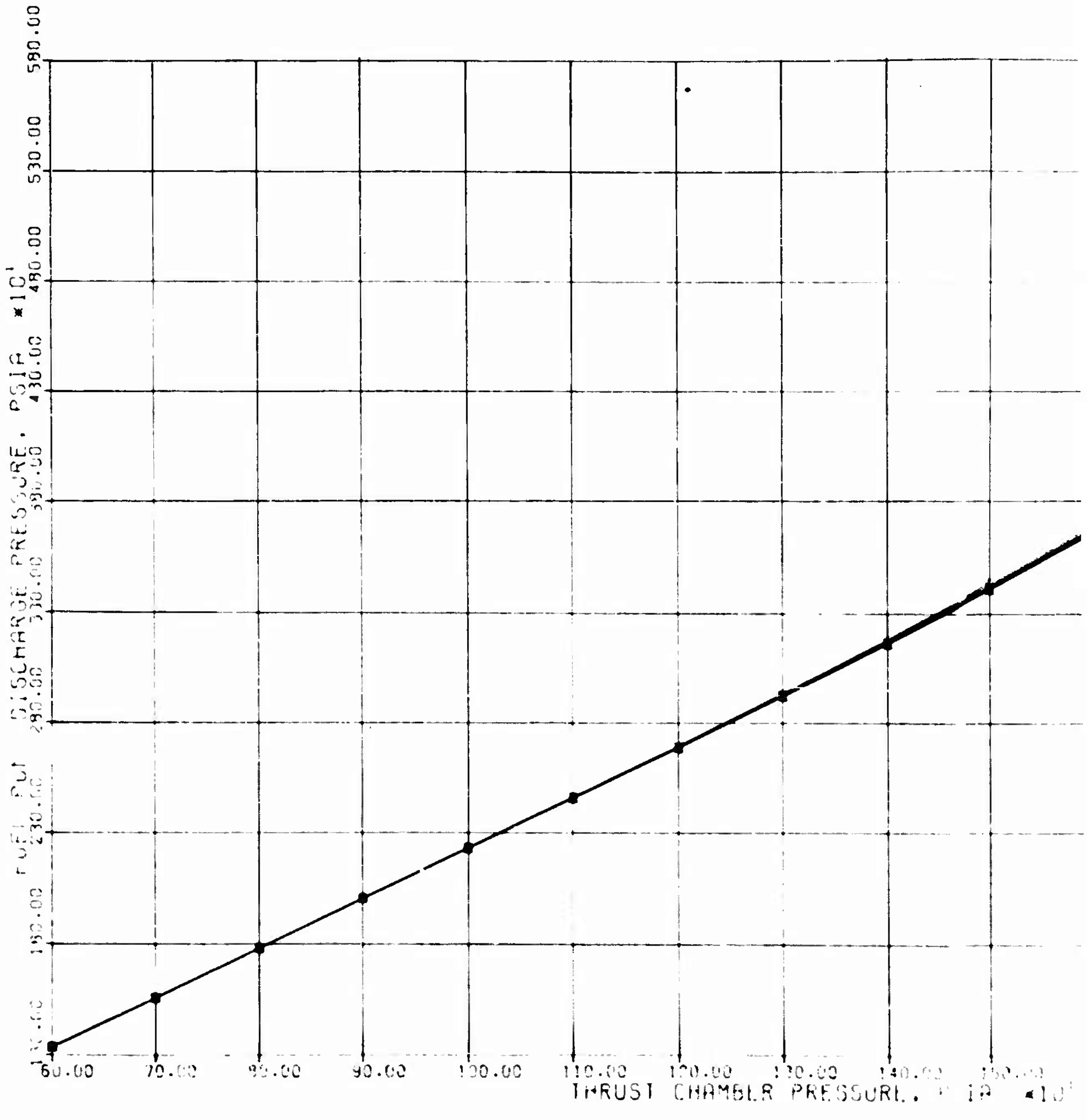
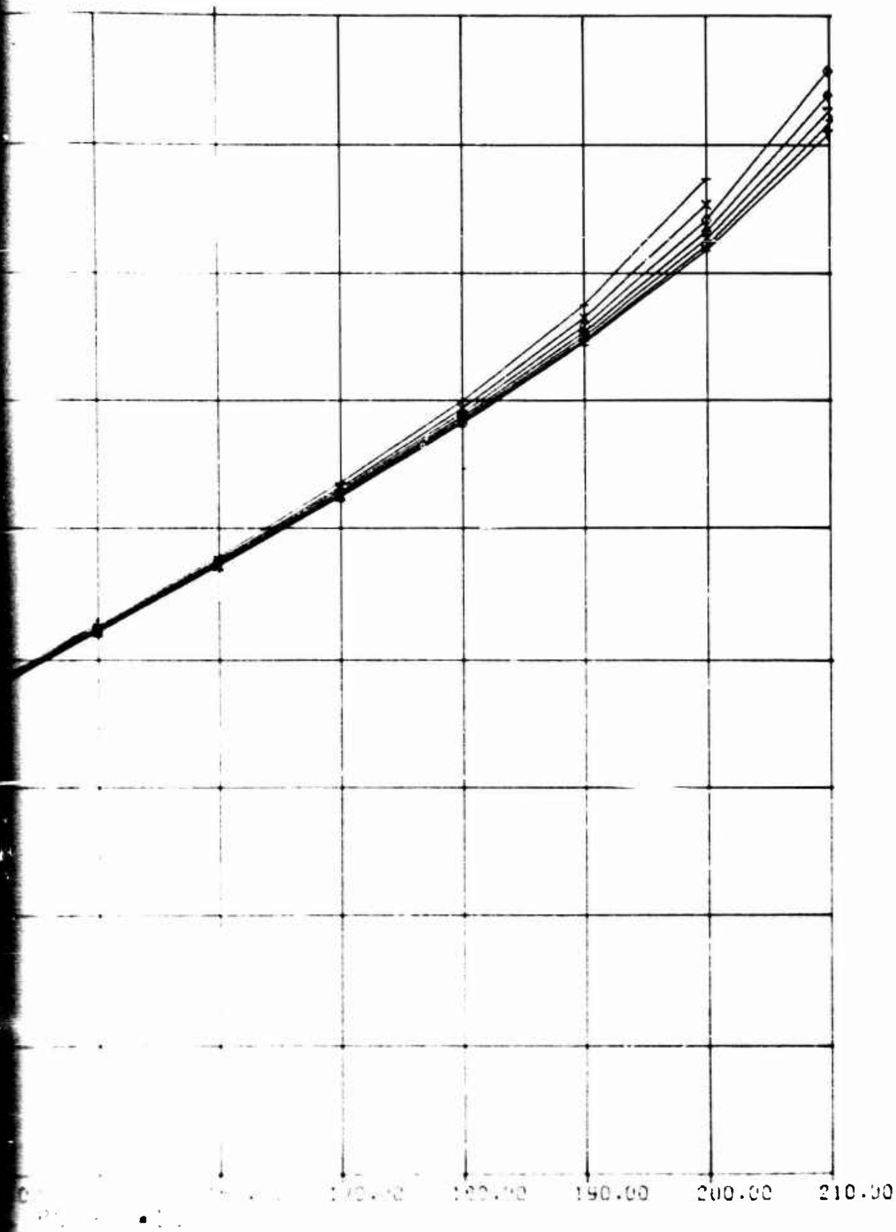


Figure 82. Power Balance f



STAGE COMBUSTION

CHAMBER MR = 6.
TTIF = 1860 °R
LH_2 PUMP STAGES - 2
TURB " - 2
LO_2 PUMP " - 1
TURB " - 1

	CHAMBER THRUST	CHAMBER PRESS. RANGE
□	8,000 LB	600 - 800 PSIA
○	10,000	600 - 1000
△	15,000	600 - 1500
+	20,000	600 - 2000
X	25,000	600 - 2000
◇	30,000	700 - 2100
ⴷ	35,000	800 - 2100
ⴳ	40,000	900 - 2100
Z	45,000	1000 - 2100
Y	50,000	1200 - 2100

...er Balance for Staged Combustion Cycle, MR = 6

f. Performance Optimization of Bleed Cycles

A chamber pressure optimization survey was conducted for the bleed cycles to obtain the optimum performance within the given envelope for both the fixed and minimum weight retractable nozzles. The relationship is indicated in the generalized performance curves shown in Figure 83.

g. Maximum Chamber Pressure Determination

In Figure 83, the maximum recommended chamber pressures are summarized for each cycle over a thrust range of 8K to 50K lb. The results indicate that the staged combustion type cycle has the highest chamber pressure capability over the whole thrust range followed by the gas generator bleed cycle. This relationship was used to determine the engine performance and payload capability of the various cycles as a function of thrust level.

8. <u>Comparison of Engine Cycles over 8K to 50K Thrust Range</u>

Engine cycles compared for specific impulse and payload capability at the design requirements.

Staged Combustion Cycle

Staged Combustion - Bleed Cycle

Expander Cycle

Gas Generator and Chamber Tap-off Cycle

Coolant Bleed Cycle

The gas generator cycle and chamber tap-off cycle are identical in weight and performance and therefore no differentiation is made between them. Both fixed and the minimum weight retractable nozzles are compared. The comparisons are valid for a mixture ratio of 6.0 for fixed and stowed nozzle envelopes over all thrust levels. The comparison of specific impulse is shown in Figure 84. The results indicate that the staged combustion cycle has the highest performance over the whole thrust range from 8K to 50K thrust for both the fixed and retractable nozzles. It also indicates an increase in performance with an increase in thrust for the retractable nozzle and a decrease in performance with increasing thrust for the fixed nozzle concept.

The staged combustion bleed cycle shows similar trends with slightly reduced performance as compared to the staged combustion cycle. The slightly reduced performance is due to the bleed losses.

The expander cycle shows optimum performance for the retractable nozzle at the 15K thrust level. The performance gain of the retractable nozzle over the fixed nozzle is considerable.

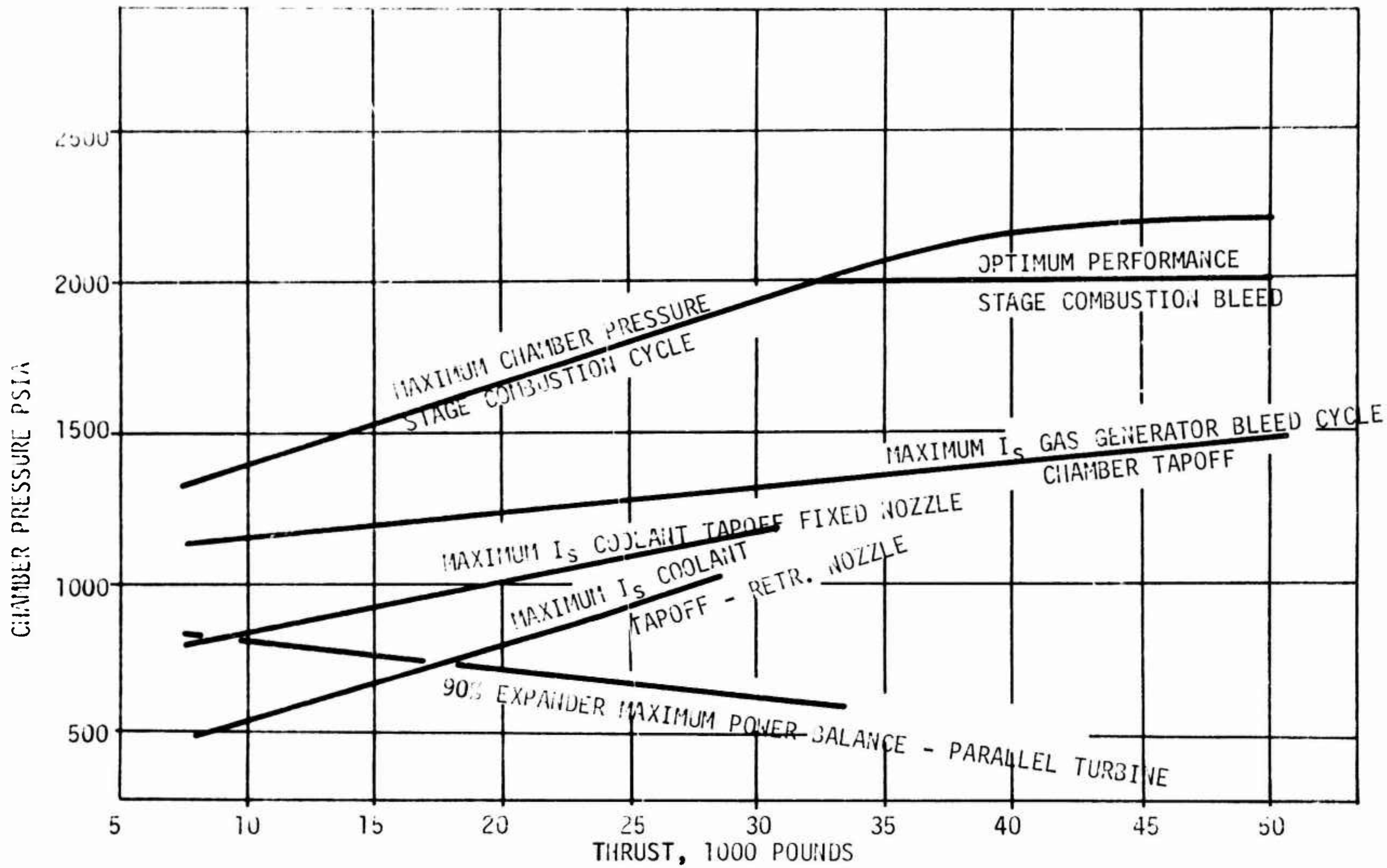


Figure 83. Optimum Chamber Pressure for Various Engine Cycles

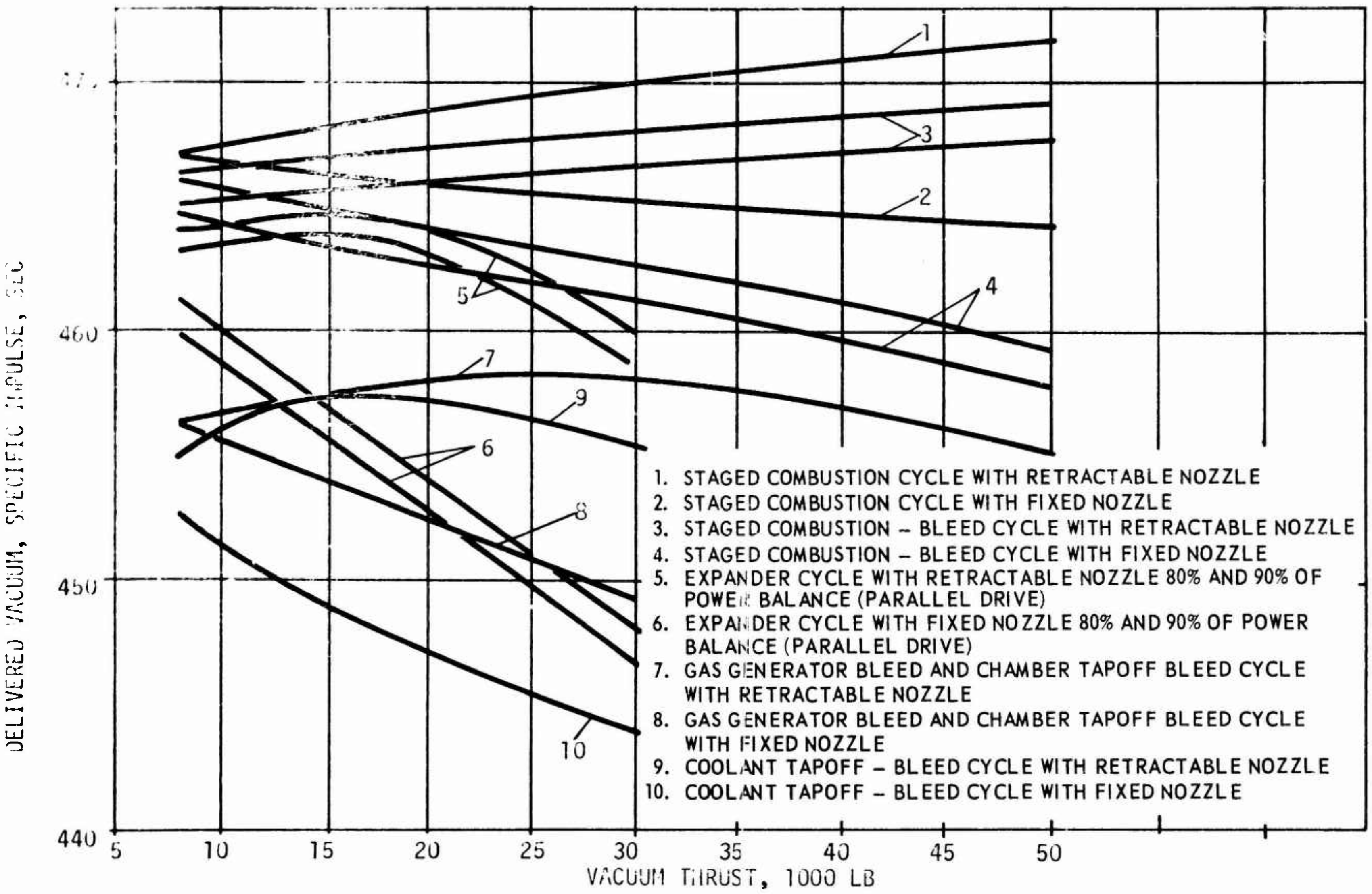


Figure 84. I_s vs Thrust for Various Engine Cycles

The gas generator bleed cycle and chamber tap-off cycle for the retractable nozzle optimizes at the 25K thrust level. It will be noted the bleed cycles can only obtain performance levels below 460 sec of specific impulse because of the relatively large bleed losses at MR = 6.0.

The coolant tap-off cycle for retractable nozzle compares favorably with the gas generator bleed cycle but shows lower performance for the fixed nozzle.

All cycles show a very small gain due to the retractable nozzle at the 8K thrust level and is due to the fact that the given envelope permits the small, fixed nozzle engines to operate at high nozzle expansion area ratio.

A further comparison was made on the basis of payload capability for the orbit-to-orbit mission and the result is presented in Figure 85. In this comparison, the relative payload gains are presented relative to the stage combustion cycle at the 25K thrust level for fixed nozzle concept. The various engine weights used for this analysis are identical to the generalized parametric engine weights for the related chamber pressures and nozzle expansion areas. All weight and performance values used are documented in Table XVI.

The payload comparison which include engine weight effects shows the same order of cycle preference as the specific impulse comparison. The staged combustion cycle gives the highest payload capability over the whole thrust range from 8K to 50K thrust. The significance of this comparison is that the smaller thrust level shows increased payload capability over the larger thrust engines but this is only true when comparing single engine applications.

At the 8K thrust level, the retractable nozzle shows no payload gain as compared to the fixed nozzle due to the fact that fixed nozzles can obtain near optimum nozzle area ratios within the given envelope.

Low pressure engines gain considerable payload at the larger thrust level compared to the high pressure cycles, such as the staged combustion cycles. Very little payload is gained with the retractable nozzle concept at any thrust level for the high pressure cycles.

9. Staged Combustion Cycle Engine Description for Discrete Thrust Levels of 8K, 15K, 25K, and 50K

This section characterizes the staged combustion cycle for discrete thrust levels. Figure 86 presents the fixed nozzle engine weight and delivered vacuum performance at mixture ratio = 6.0 in generalized form as a function of thrust chamber pressure and nozzle expansion area ratio. Figure 87 shows identical data for the minimum weight retractable nozzle.

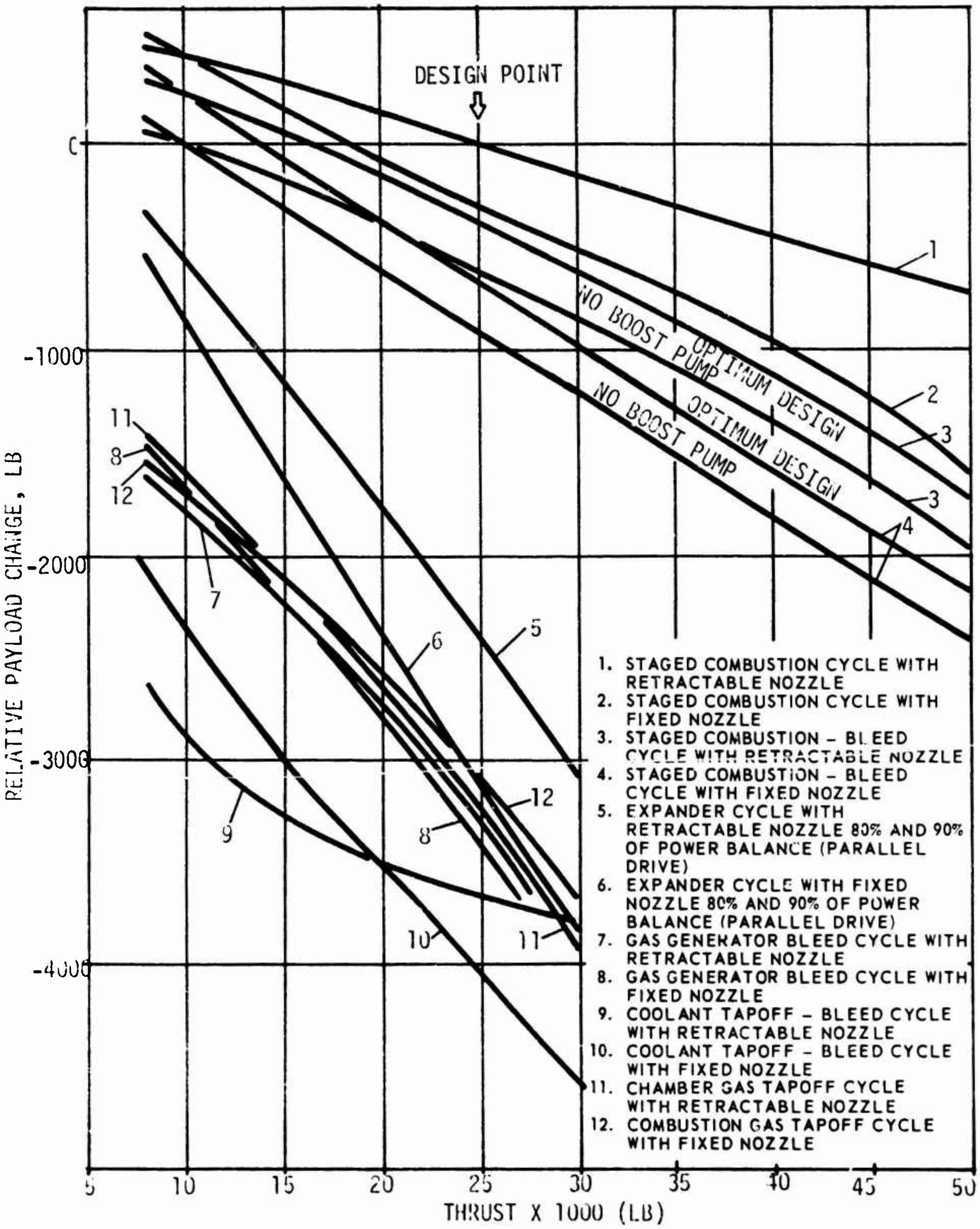


Figure 85. Relative Payload vs Thrust for Various Engine Cycles

TABLE XVI

PARAMETRIC STUDY, PAYLOAD OPTIMIZATION

MR = 6.0
Nom. NPSH(F) = 60 ft.
NPSH(O) = 16 ft.

ENGINE CYCLE Staged Combustion - Parallel Turbine Drive

FIXED NOZZLE CONFIGURATION

[illegible]	8,000	15,000	25,000	40,000	50,000
[illegible]	500	360	270	205	180
P_c	1,200	1450	1800	2140	2200
I_s	466.5	465.8	465.3	463.5	461.6
$W_{B.0}$ (Lbs)	200.0	270.0	380.0	580	660.0
157 I_s	73306	150	73052	72800	550
-3.68 $W_{B.0}$	736	993	1398	2134	2430
Relative PAYLOAD (PL) (Lbs)	72564	72157	71654	70676	69920

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

[illegible]	8,000	15,000	25,000	40,000	50,000
[illegible]	500	500	500	500	500
[illegible]	1200	1450	1800	2140	2200
[illegible]	466.6	468.3	469.1	471.0	471.4
[illegible]	230.0	330.0	490.0	730.0	940
[illegible]	73319	73523	73650	73947	74073
[illegible]	847	1214	1803	8684	3460
[illegible] (PL)	72472	72309	71847	71863	70613

ENGINE CYCLE Staged Combustion - Bleed
Series Turbine Drive

MR = 6.0
Nom NPSH(F) = 60 ft.
NPSH(O) = 16 ft.

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	25,000	40,000	50,000
ε_o	500	360	270	200	170
P_c	1200	1500	1800	2000	2000
I_s Optimum	466.1	465.0	463.5	461.0	459.2
I_s Delivered	464.6	463.5	462.0	459.5	457.7
$W_{B.O}$ (Lbs)	200.0	280.0	400.0	620.0	700.0
157 I_s Opt.	73178	73005	72770	72377	72094
157 I_s					
-3.68 $W_{B.O}$	736	1030	1472	2282	2580
PAYLOAD Opt.	72465	72015	71423	70511	69923
PAYLOAD (PL) (Lbs)	72442	71975	71298	70095	69514

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	25,000	40,000	50,000
ε_o	500	500	500	500	460
P_c	1200	1500	1800	2000	2000
I_s Optimum	466.1	467.0	467.7	468.5	468.5
I_s Delivered	464.6	465.5	466.2	467.0	467.0
$W_{B.O}$ (Lbs)	235.0	350.0	510.0	740.0	930.0
157 I_s Opt.	73178	73319	73429	73555	73555
157 I_s					
-3.68 $W_{B.O}$	864	1250	1877	2720	3420
PAYLOAD Opt.	72392	72149	71708	70986	70373
PAYLOAD (PL) (Lbs)	72314	72069	71552	70735	70135

MR = 6.0
Nom. NPSH(F) = 60 ft
NPSH(O) = 16 ft.

ENGINE CYCLE — Expander - Parallel Turbine Drive

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	25,000	30,000	50,000
ε_e	300	180	100	N/A	N/A
P_c	820	760	680	660	-
I_s	461.2	456.8	451.2	Weight Excessive	Increase
$W_{B.O}$ (Lbs)	227.9	343.5	507.2	——	——
157 I_s	72408	71718	70838	——	——
-3.68 $W_{B.O}$	839	1265	1866	——	——
Relative PAYLOAD (PL) (Lbs)	71569	70453	68972	——	——

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
[illegible]	500	500	400	320	250
[illegible]	820	760	710	680	660
[illegible]	464.0	464.6	464.0	462.4	460.2
[illegible].O (Lbs)	293.5	545.4	678.8	789.8	878.4
[illegible]	72848	72942	72848	72597	72251
-3.68 $W_{B.O}$	1080	2007	2498	2906	3233
[illegible] (PL)	71768	70935	70350	69691	69018

TABLE XVI (cont.)

ENGINE CYCLE Gas Generator Bleed - Series Turbine Drive

MR = 6.0
Nominal NPSH(F) = 60 Ft.
NPSH(O) = 16 Ft.

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
ε_o	500	295	235	205
P_c	1130	1190	1225	1270
I_s	455.6	454.0	453.0	451.6
$W_{B.O}$ (Lbs)	230.0	340.0	420.0	500.0
157 I_s	71550	71278	71100	70900
-3.68 $W_{B.O}$	846	1250	1545	1840
Relative PAYLOAD (PL) (Lbs)	70764	70028	69555	69.060

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
ε_o	500	500	500	500
P_c	1130	1190	1225	1270
I_s	455.6	457.6	458.7	459.5
$W_{B.O}$ (Lbs)	260.0	410.0	550.0	680.0
157 I_s	71400	71812	72000	72080
-3.68 $W_{B.O}$	957	1508	2025	2503
Relative PAYLOAD (PL) (Lbs)	70443	70304	69975	69577

TABLE XVI (cont.)

MR = 6.0
Nom. NPSH(F) = 60 ft
NPSH(O) = 16 ft.

ENGINE CYCLE Coolant Tapoff - Series Turbine Drive

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
ε_o	360	240	200	180	165
P_c	800	930	1000	1090	1170
I_s	453.4	450.2	448.5	446.8	445.4
$W_{B.O}$ (Lbs)	230.0	330.0	400.0	47.0	Area Ratio ε_o Limited
157 I_s	71200	70700	70400	70100	
-3.68 $W_{B.O}$	846	1215	1473	1730	———
Relative PAYLOAD (PL) (Lbs)	70354	69485	68927	68370	———

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
ε_o	500	500	460	430	405
P_c	500	680	800	930	1060
I_s	454.5	458.5	458.4	457.7	456.7
$W_{B.O}$ (Lbs)	390.0	570.0	630.0	700.0	Weight Excessive
157 I_s	71400	72000	72000	71820	
-3.68 $W_{B.O}$	1435	2100	2320	2580	———
Relative PAYLOAD (PL) (Lbs)	69965	69900	69680	69240	———

TABLE XVI (cont.)

ENGINE CYCLE — Staged Combustion - Bleed, Series Turbine Drive

MR = 6.0
Nom NPSH(F) = 60 ft.
NPSH(O) = 16 ft.

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	25,000	40,000	50,000
ε_o	500	360	270	200	170
P_c	1300	1500	1800	2000	2000
I_s Optimum	466.1	465.0	463.5	461.0	459.2
I_s Delivered	464.6	463.5	462.0	459.5	457.7
$W_{B.O}$ (Lbs)	193.7	269.0	366.1	507.0	590.0
157 I_s Opt.	73178	73005	72770	72377	72094
157 I_s	72942	72770	72534	72142	71859
-3.68 $W_{B.O}$	713.	990.	1347.	1866.	2171.
Relative PAYLOAD Opt.	72465	72015	71423	70511	69923
PAYLOAD (PL) (Lbs)	72229	71780	71187	70276	69688

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	25,000	40,000	50,000
ε_o	500	500	500	500	460
P_c	1300	1500	1800	2000	2000
I_s Optimum	466.1	467.0	467.7	468.5	468.5
I_s Delivered	464.6	465.5	466.2	467.0	467.0
$W_{B.O}$ (Lbs)	213.6	318.0	467.6	698.0	864.6
157 I_s Opt.	73178	73319	73429	73555	73555
157 I_s	72942	73084	73193	73319	73319
-3.68 $W_{B.O}$	786	1170	1721	2569	3182
Relative PAYLOAD (Opt.	72392	72149	71708	70986	70373
PAYLOAD (PL) (Lbs)	72156	71914	71472	70750	70137

TABLE XVI (cont.)

ENGINE CYCLE Expander - Parallel Turbine Drive

MR = 6.0
Nom. NPSH(F) = 60 ft.
NPSH(O) = 16 ft.

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	25,000	30,000	50,000
ε	300	180	100	N/A	N/A
P_c	820	760	680	660	-
I_s	461.2	456.8	451.2	Weight Increase Excessive	
$W_{B.O}$ (Lbs)	227.9	343.5	507.2	—	—
157 I_s	72408	71718	70838	—	—
$-3.68\ W_{B.O}$	839	1265	1866	—	—
Relative PAYLOAD (PL) (Lbs)	71569	70453	68972	—	—

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
ε	500	500	400	320	250
P_c	820	760	710	680	660
I_s	464.0	464.6	464.0	462.4	460.2
$W_{B.O}$ (Lbs)	293.5	545.4	678.8	789.8	878.4
157 I_s	72848	72942	72848	72597	72251
$-3.68\ W_{B.O}$	1080	2007	2498	2906	3233
Relative PAYLOAD (PL) (Lbs)	71768	70935	70350	69691	69018

TABLE XVI (cont.)

MR = 6.0
Nominal NPSH(F) = 60 Ft.
NPSH(O) = 16 Ft.

ENGINE CYCLE   Gas Generator Bleed - Series Turbine Drive

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
ε_o	500	295	235	205
P_c	1130	1190	1225	1270
I_s	456.3	454.0	452.4	450.4
$W_{B.O}$ (Lbs)	270.9	390.6	470.6	549.4
157 I_s	71639	71278	71027	70713
-3.68 $W_{B.O}$	997.	1437.	1732.	2022
Relative PAYLOAD (PL) (Lbs)	70642	69841	69295	68691

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
ε_o	500	500	500	500
P_c	1130	1190	1225	1270
I_s	456.3	457.4	458.1	458.3
$W_{B.O}$ (Lbs)	317.1	538.5	697.1	860.7
157 I_s	71639	71812	71922	71953
-3.68 $W_{B.O}$	1167.	1982.	2565.	3167.
Relative PAYLOAD (PL) (Lbs)	70472	69830	69357	68786

TABLE XVI (cont.)

MR = 6.0
Nom. NPSH(F) = 60 ft
NPSH(O) = 16 ft

ENGINE CYCLE Combustion Gas Tapoff - Series Turbine Drive

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
ε_o	500	295	235	205
P_c	1130	1190	1225	1270
I_s	456.3	454.0	452.4	450.4
$W_{E.O}$ (Lbs)	249.5	356.2	426.5	500.0
157 I_s	71639	71278	71027	70713
-3.68 $W_{B.O}$	918	1311	1570	1840
Relative PAYLOAD (PL) (Lbs)	70721	69967	69457	68873

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000
[illegible]	500	500	500	500
[illegible]	1130	1190	1225	1270
[illegible]	456.3	457.4	458.1	458.3
[illegible]	295.6	503.9	655.0	807.4
[illegible]	71639	71812	71922	71953
[illegible]	1088	1854	2410	2971
[illegible]	70551	69958	69512	68982

TABLE XVI (cont.)

MR = 6.0
Nom. NPSH(F) = 60 f
NPSH(O) = 16 f

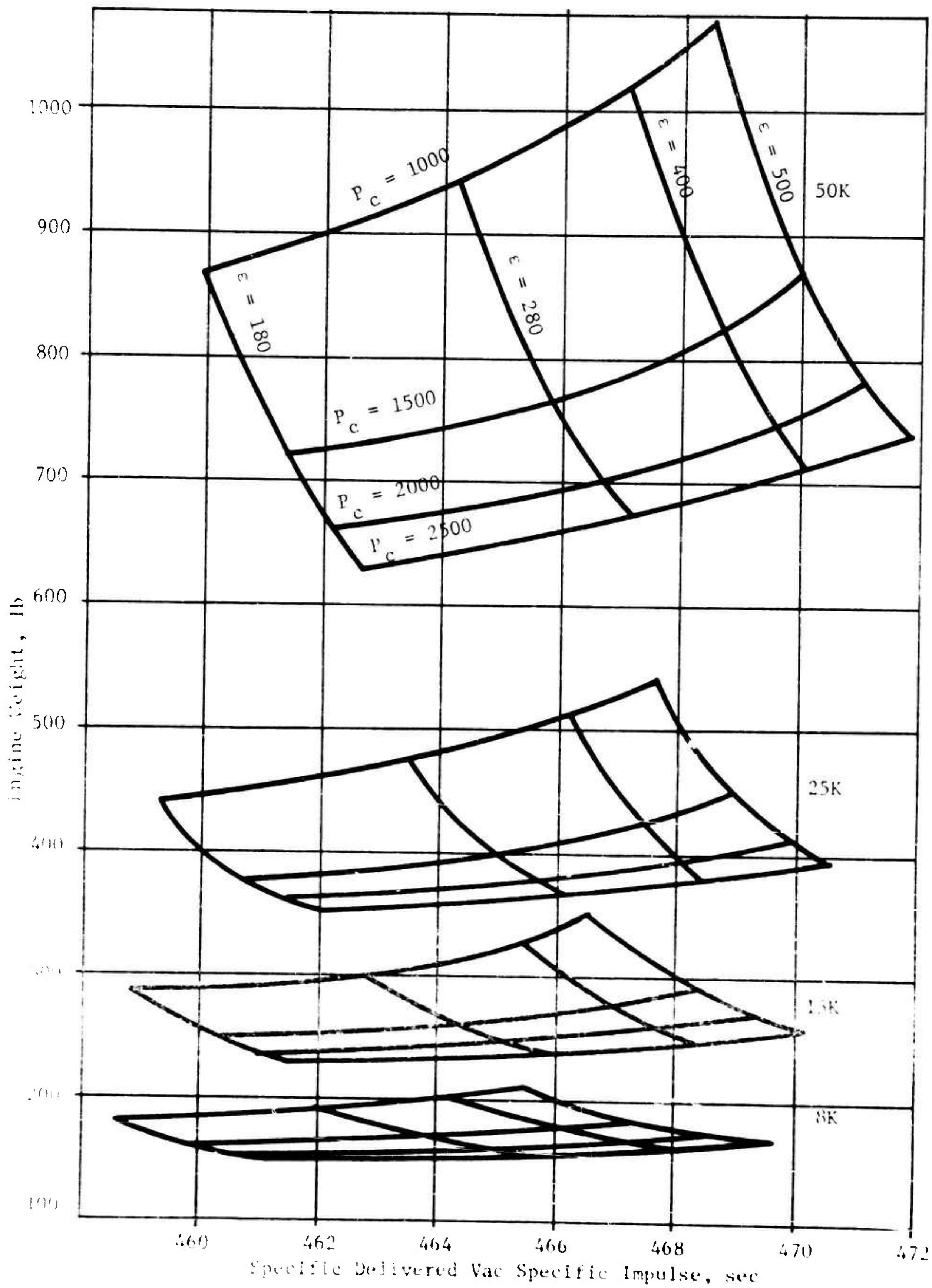
ENGINE CYCLE Coolant Tapoff - Series Turbine Drive

FIXED NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
ϵ_o	360	240	200	180	165
P_c	800	930	1000	1090	1170
I_s	452.6	449.1	447.3	445.5	444.0
$W_{B.O}$ (Lbs)	283.1	387.6	448.4	518.5	Area Ratio ϵ_o Limited
157 I_s	71058	70509	70226	69944	69708
-3.68 $W_{B.O}$	1042	1426	1650	1908	———
Relative PAYLOAD (PL) (Lbs)	70016	69083	68576	68036	———

MINIMUM WEIGHT RETRACTABLE NOZZLE CONFIGURATION

F	8,000	15,000	20,000	25,000	30,000
ϵ_o	500	500	460	430	405
P_c	500	680	800	930	1060
I_s	454.7	457.4	457.2	456.4	455.3
$W_{B.O}$ (Lbs)	523.7	816.3	912.0	945.0	Weight Excessive
157 I_s	71388	71812	71780	71655	71482
-3.68 $W_{B.O}$	1927	3004	3356	3478	———
Relative PAYLOAD (PL) (Lbs)	69461	68808	68424	68177	———



. ... Combustion Cycle Engine Weight and I_s, MR = 6, Fixed Nozzle

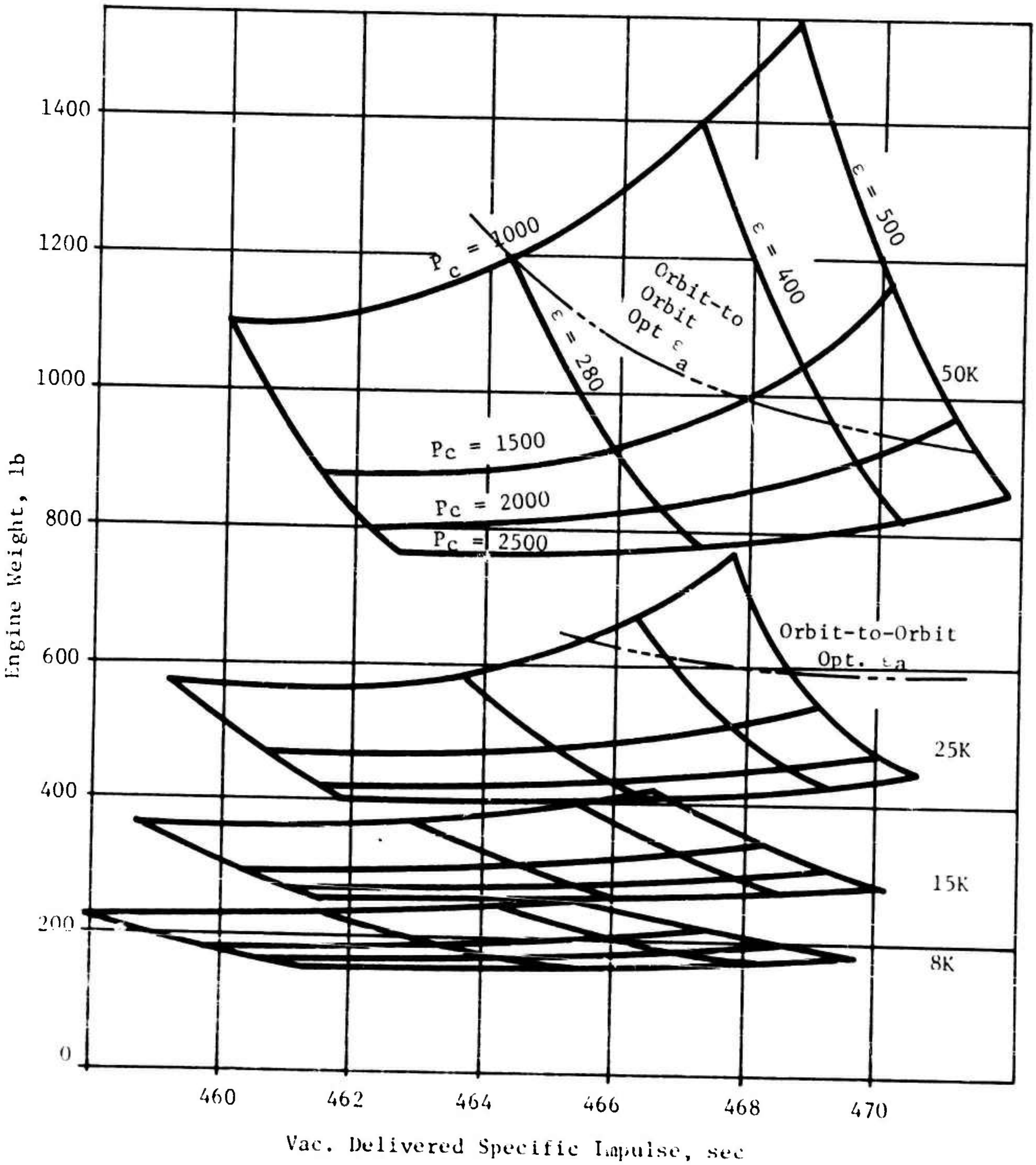


Figure 87. Staged Combustion Cycle Engine Weight and I_s, MR = 6, Minimum Weight Retractable Nozzle

The engine configuration is identical to that used in determining the parametric data with minimum length Rao nozzle and a thrust chamber length of 8 inches. The injector energy release efficiency is assumed to be 99% and the performance is calculated with the JANNAF performance prediction methods. Suction NPSH is 16 ft for the LO_2 pump and NPSH = 60 ft for the hydrogen pump.

In Table XVII, the engines are characterized for the optimum chamber pressures meeting the thermal cycle requirements of N_t = 300 cycles and conforms to the stowed engine length of 82 inches.

The turbine drive cycles are presented for the four discrete thrust levels in Figures 88 through 91 containing the flow pressure and temperature schedule for nominal conditions.

The following table describes the feed system selected for each thrust level.

TURBOPUMP - FUEL CIRCUIT
Engine MR = 6 NPSH = 60 feet

	Thrust			
	8K	15K	25K	50K*
Boost Pump				
Shaft Speed, rpm	31428	25714	22857	20000
Suction Specific Speed	9816	10996	12622	15675
Main Pump				
Number of Stages	2	2	3	3
Shaft Speed, rpm	11000	90000	80000	70000
Impeller Tip Speed, ft/sec	1755	1682	1502	1621
Specific Speed	504	514	724	841
Efficiency, %	51.7	54.7	61.6	66.0
Discharge Pressure, psia	3279	3721	4260	4682
Volume Flow, gpm	254	476	794	1598
Turbine				
Inlet Temperature, °R	1860	1860	1860	1860
Mean Blade Speed, ft/sec	1300	1300	1300	1300
Number Stages	2	2	2	2
Nozzle Admission, %	100	100	100	100
Efficiency, %	66.9	66.1	65.7	66.9
Shaft Power, hp	979	1970	3343	7041

[illegible]: 23 Sept 71 22:02:22.593
24 Sept 71 16:10:03.076

TABLE XVII

STAGED COMBUSTION CYCLE SUMMARY VS THRUST LEVEL

	Fixed Nozzle				Min Height Retractable			
Thrust, lb	8	15	25	50	8	15	25	50
Nozzle Area Ratio	500	360	270	180	500	500	500	500
Chamber Pressure, psia	1200	1450	1800	2200	1200	1450	1800	2200
Operating Engine Length, in.	82	82	82	82	-	98	110	136
Max Diameter, in.	47.5	49.5	50.5	53.0	-	60	68	87
Engine Weight, in.	198	270	380	660	-	350	490	940
Suction Line Dia Fuel, in.	2.2	2.9	3.60	4.90	2.2	2.9	3.60	4.9
Suction Line Dia LOX, in.	1.9	2.5	3.1	4.3	1.9	2.5	3.1	4.3
F/Pc	6.60	10.33	13.88	22.70	6.6	10.33	13.88	22.70
Specific Impulse, sec	466.5	465.8	465.3	461.6	-	468.3	469.8	471.20
NPSH Fuel, ft	60	60	60	60	60	60	60	60
NPSH LOX, ft	16	16	16	16	16	16	16	16

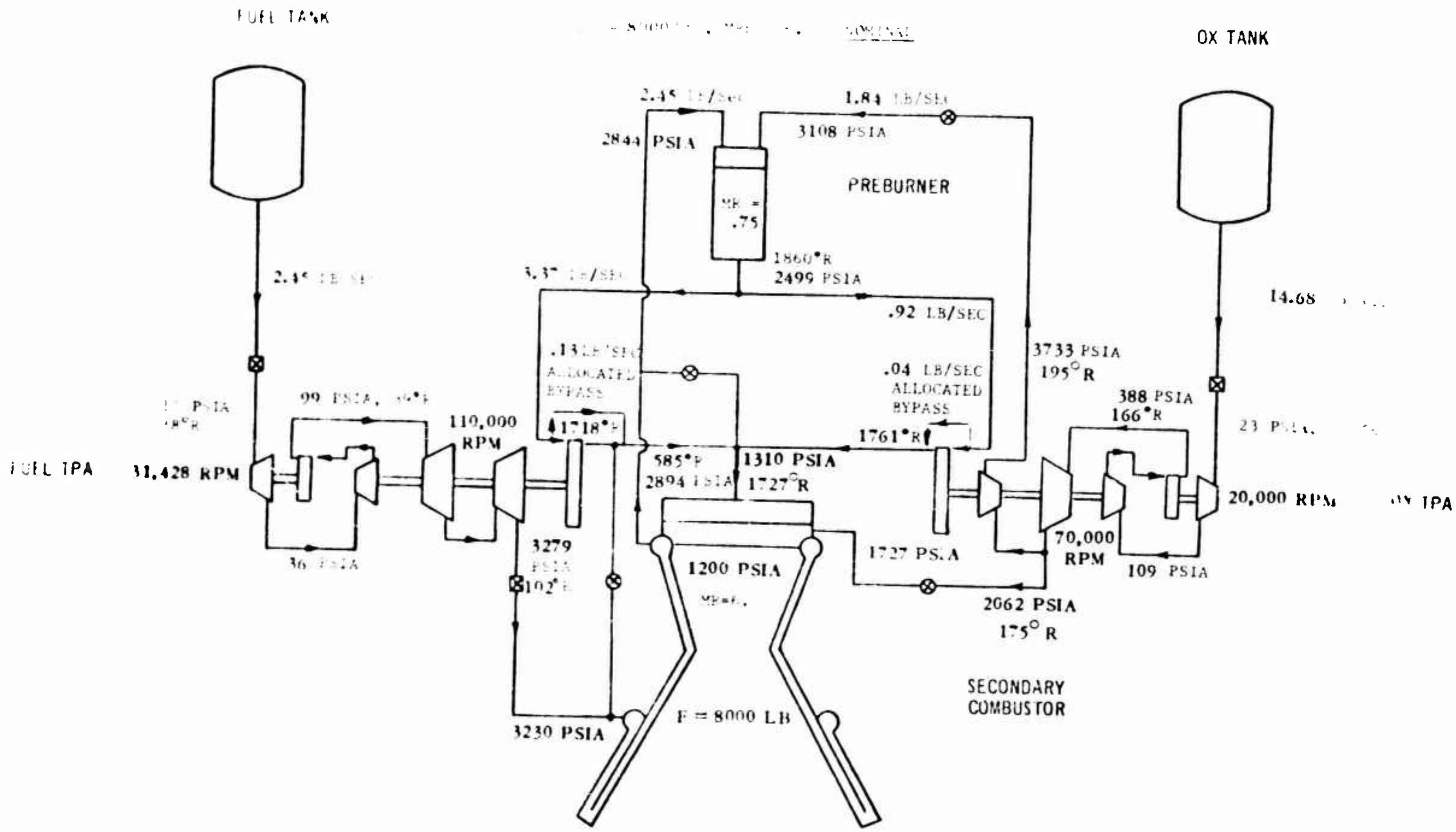


Figure 88. Parameters for Engine Operation at Full Thrust, F = 8

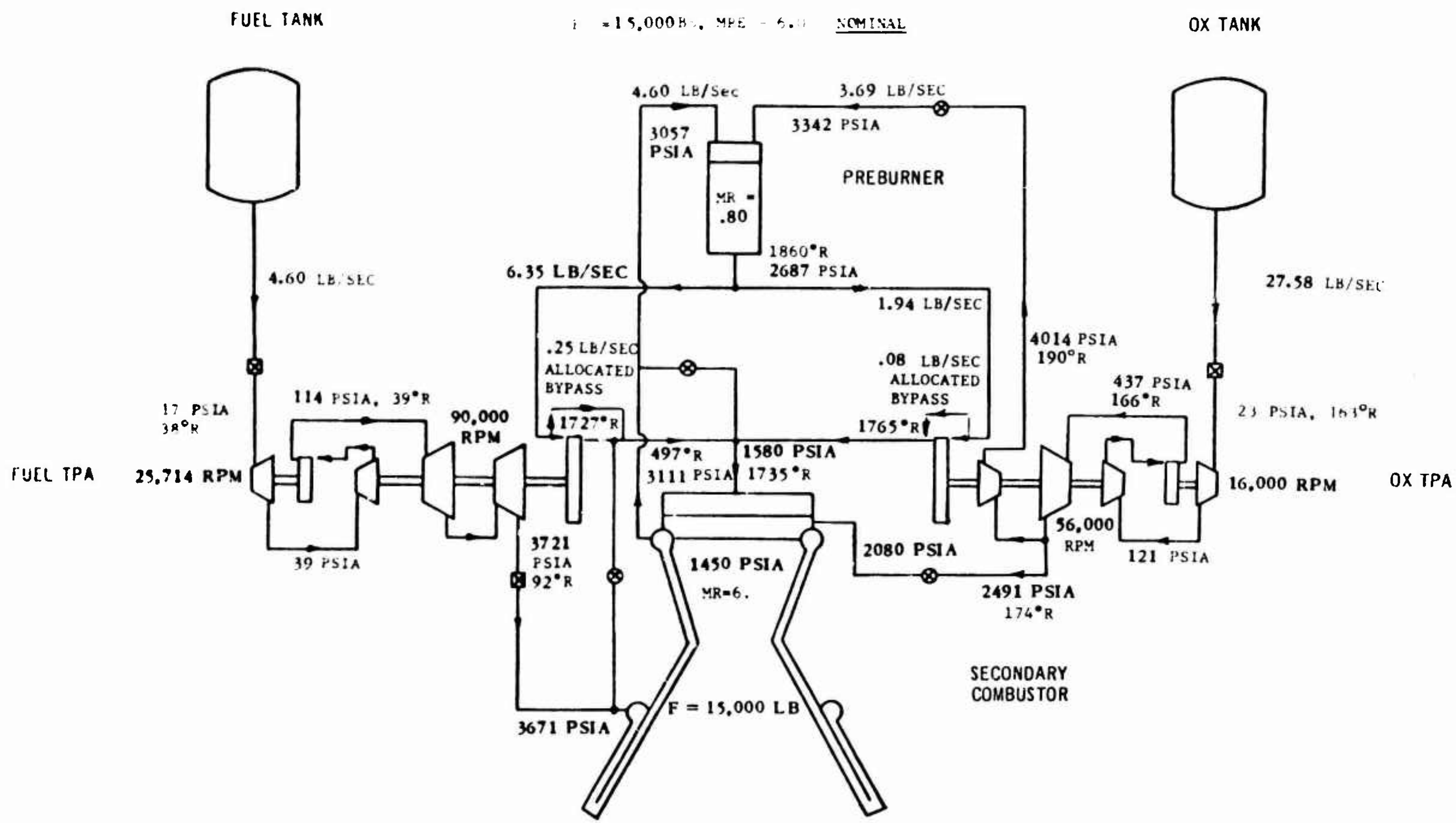


Figure 89. Parameters for Engine Operation at Full Thrust, F = 15K

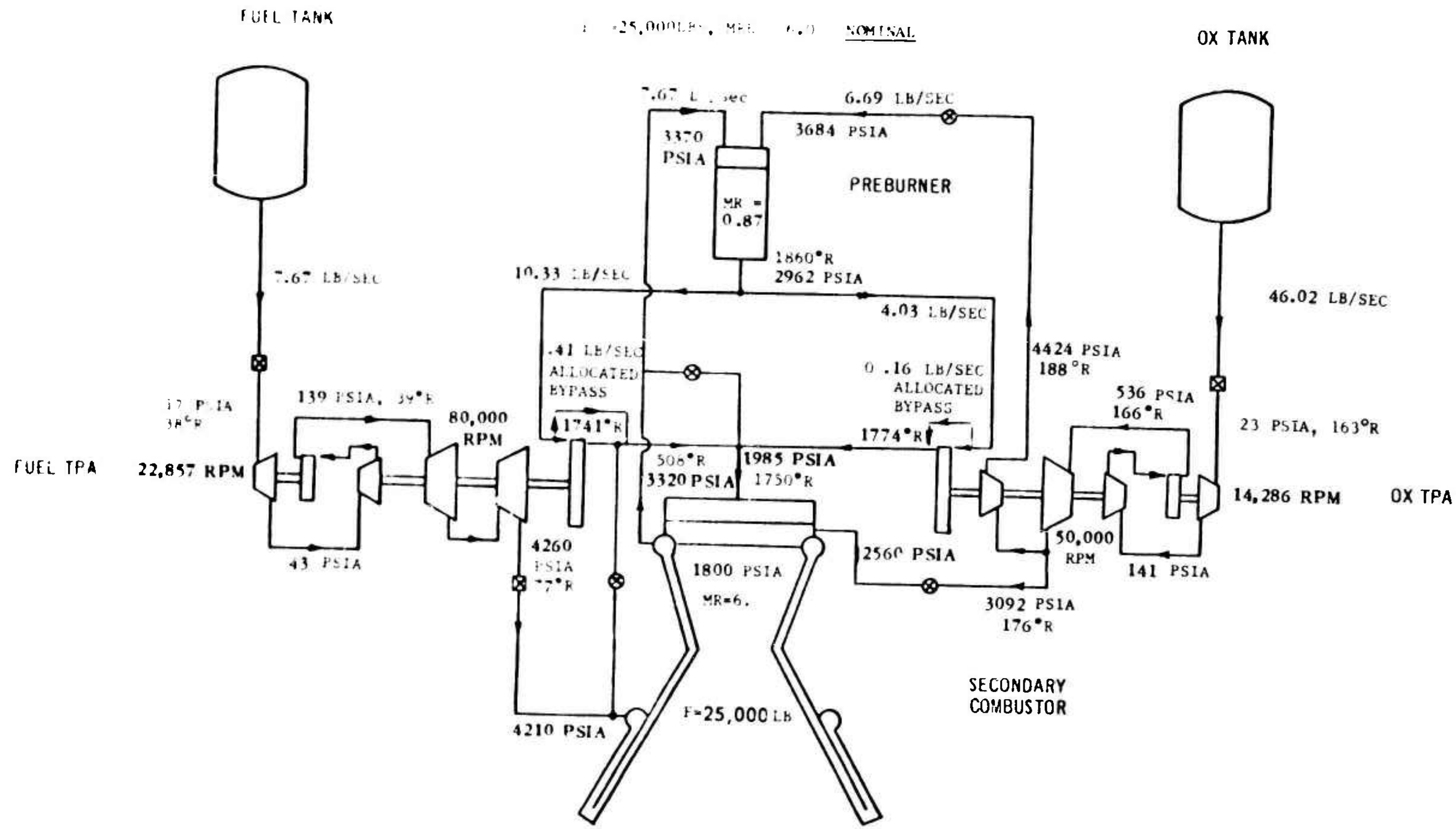


Figure 90. Parameters for Engine Operation at Full Thrust, F = 25K

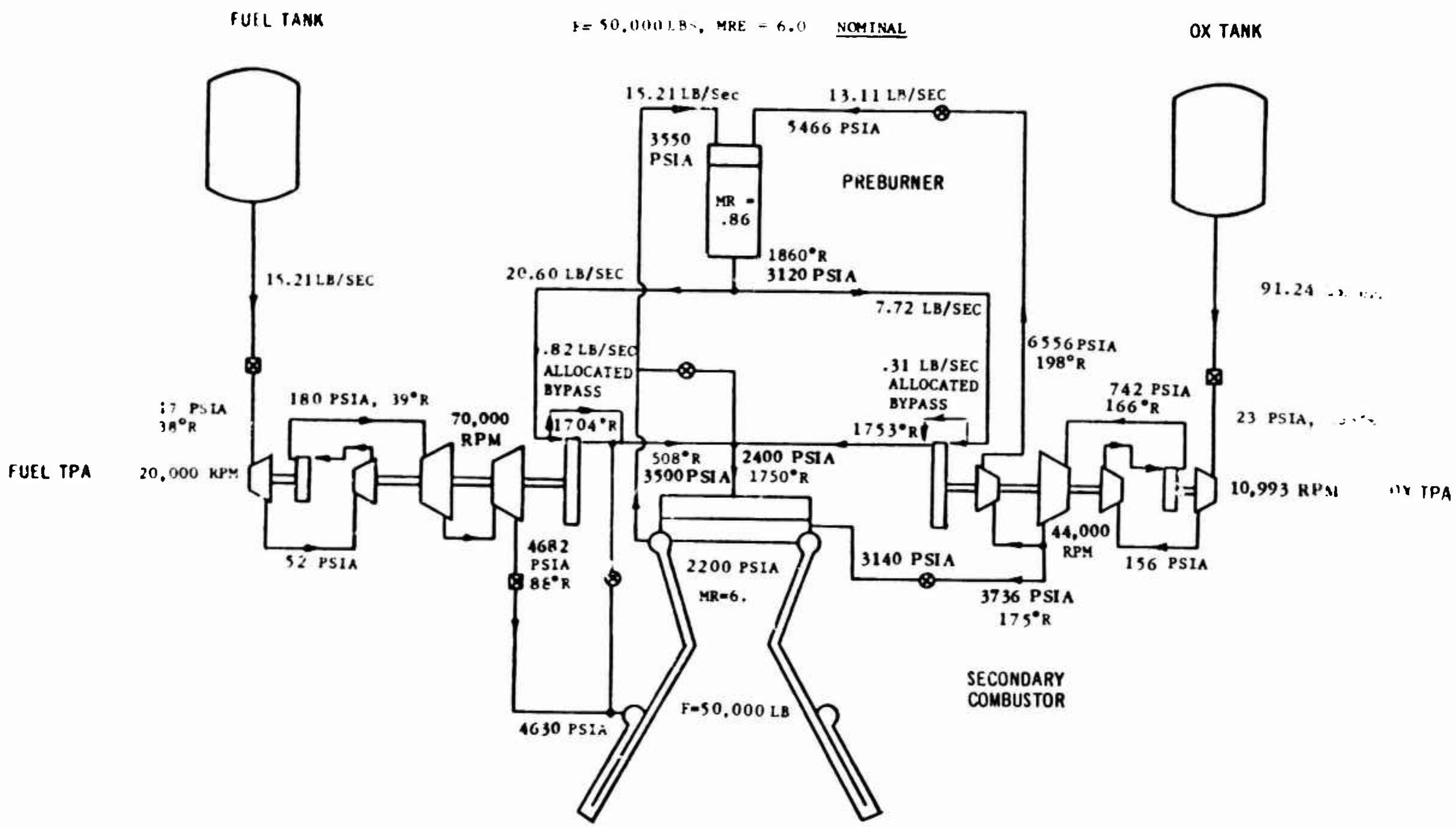


Figure 91. Parameters for Engine Operation at Full Thrust, F = 50K

TURBOPUMP - OXIDIZER CIRCUIT
Engine MR = 6 NPSH = 16 Feet

	Thrust			
	8K	15K	25K	50K**
Boost Pump				
Shaft Speed, rpm	20000	16000	14286	10993
Suction Specific Speed	20404	22350	25768	28140
Main Pump				
Number Stages	1-1/2	1-1/2	1-1/2	1-1/2
Shaft Speed, rpm	70000	56000	50000	44000
Impeller Tip Speed, ft/sec	488	533	591	634
Specific Speed	1522	1430	1399	1683
Efficiency, %	57.6	61.1	63.8	68.6
Discharge Pressure*, psia	2062	2491	3092	3736
Volume Flow, gpm	93.1	175	291	586
Turbine				
Inlet Temperature, °R	1860	1860	1860	1860
Mean Blade Speed, ft/sec	1100	1100	1100	1100
Number Stages	1	1	1	1
Nozzle Admission, %	28	27.5	32.3	44.3
Efficiency, %	48.8	48.0	47.6	47.7
Shaft Power, hp	228	487	972	1986

*Main Stage
**Ref Run: 23 Sept 71 22:02:22.593
24 Sept 71 16:10:03.076

10. Mixture Ratio and NPSH Data for 8K, 15K and 25K Thrust

This section describes the effect of varying design mixture ratio and NPSH on the staged combustion cycle for three different thrust levels.

Design mixture ratio effect exerts a strong influence on engine performance and to a lesser extent on the engine weight and envelope. The effects are presented for the fixed and the minimum weight retractable nozzle.

Figures 92 and 93 present generalized data for engine weight and performance for MR = 5 and MR = 7.0 as a function of thrust, chamber pressure and nozzle expansion area ratio.

In addition, engine performances for MR = 5.0 and MR = 7.0 are shown separately for each thrust level in Figures 94 through 99.

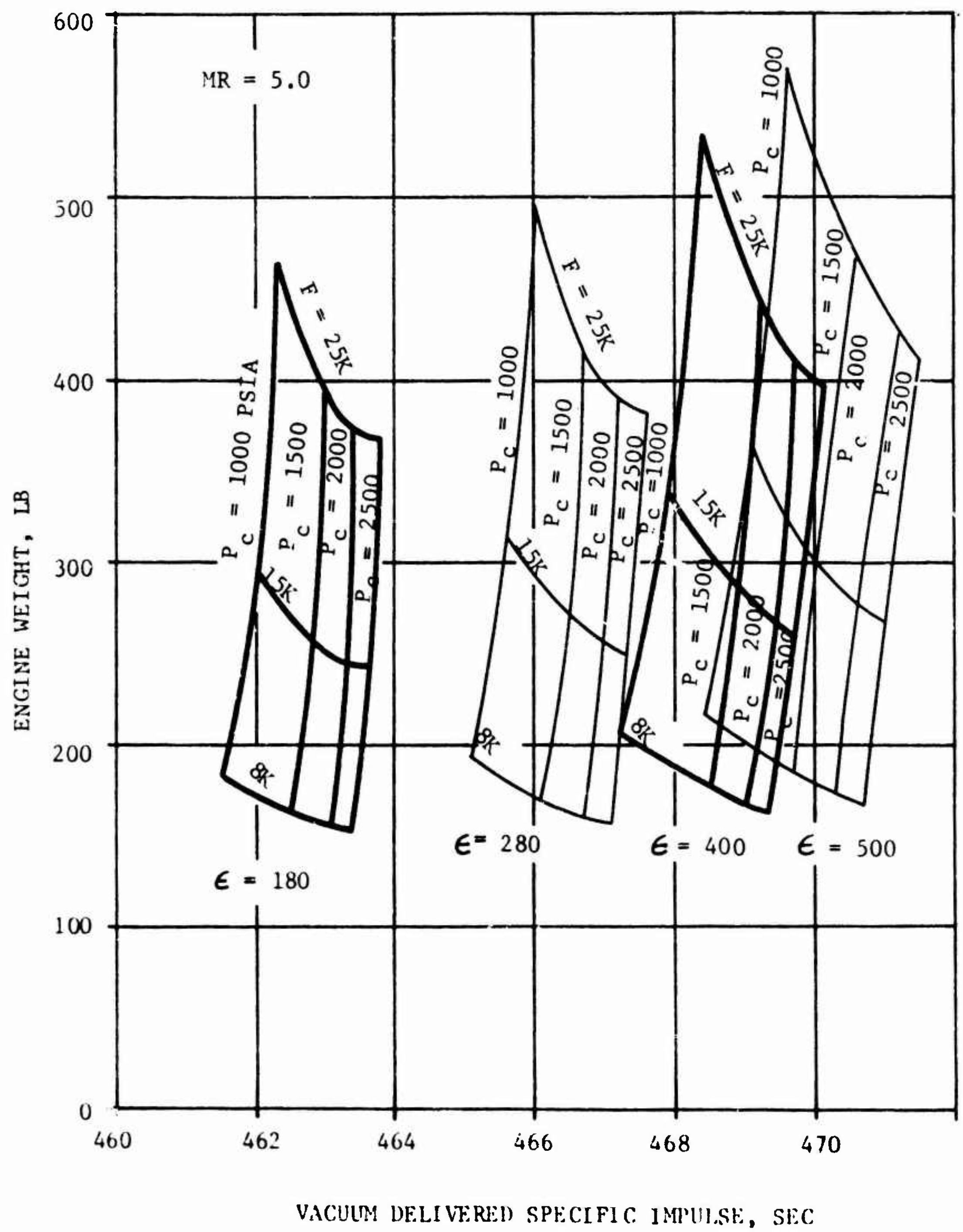


Figure 92. Generalized Weight and Performance Data, MR = 7, Staged Combustion Cycle

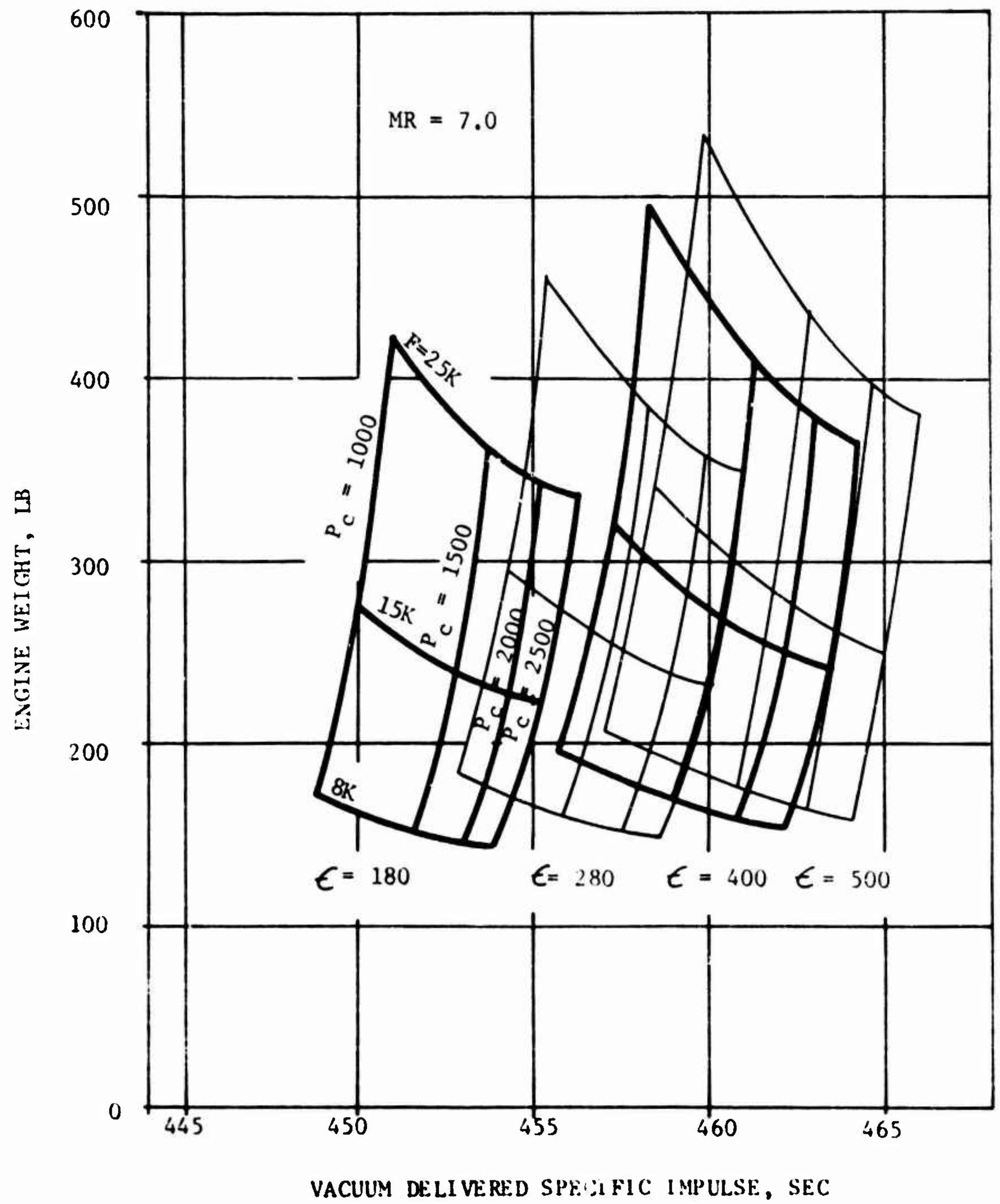


Figure 93. Generalized Weight and Performance Data, MR = 5, Staged Combustion Cycle

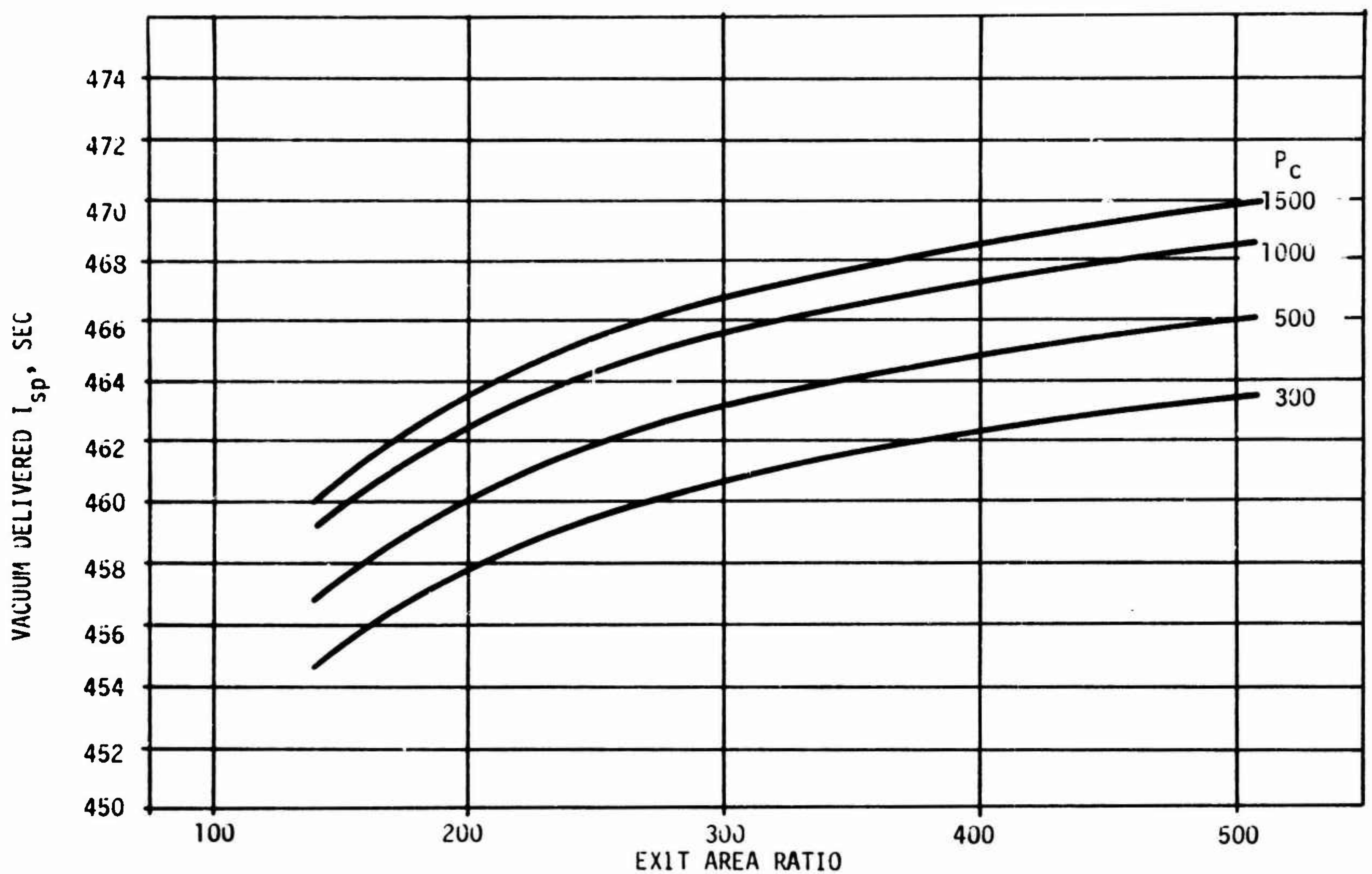


Figure 94. I_s vs Exit Area Ratio, F = 8K, MR = 5, Staged Combustion Cycle

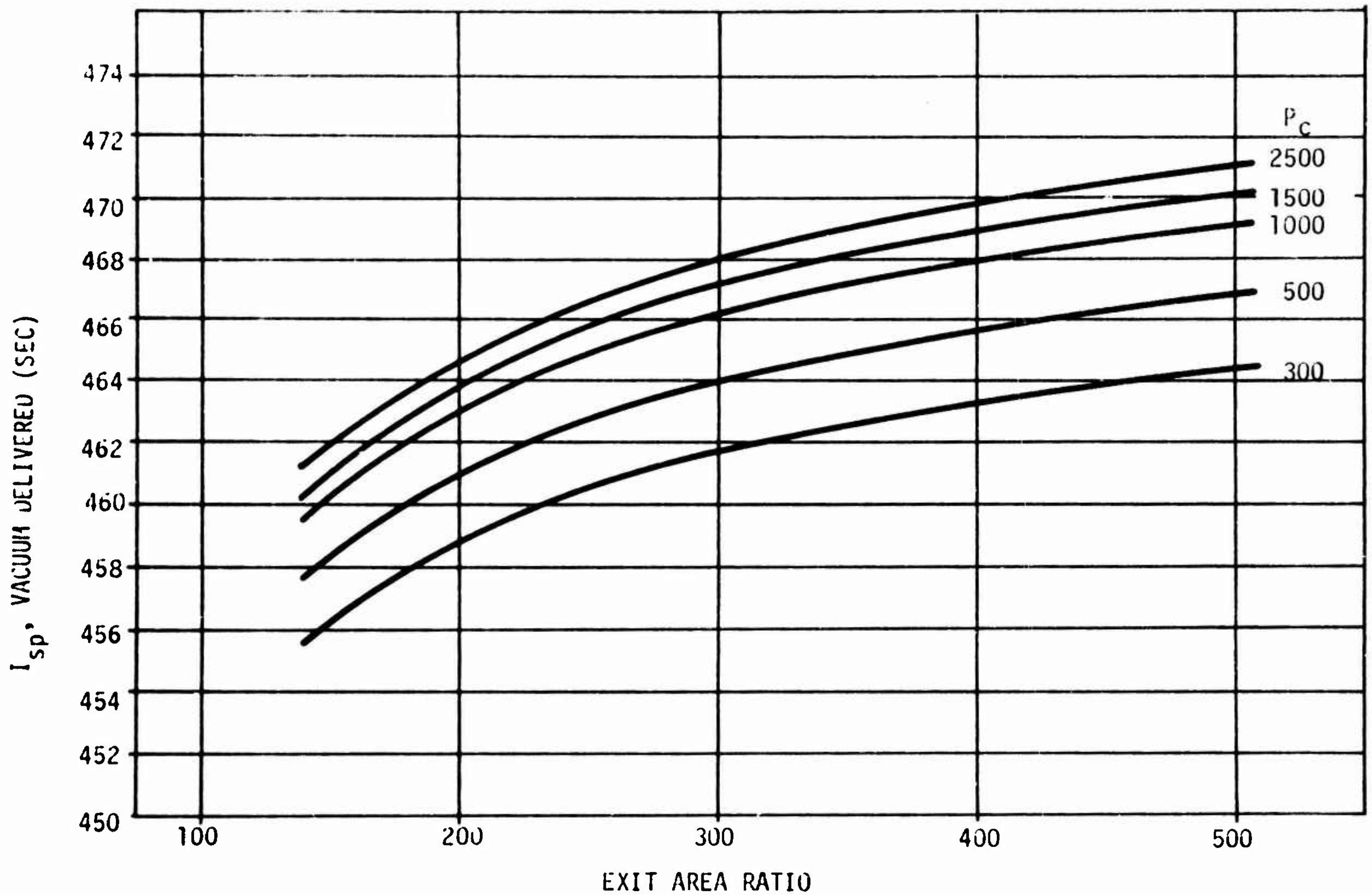


Figure 95. I_s vs Exit Area Ratio, F = 15K, MR = 5, Staged Combustion Cycle

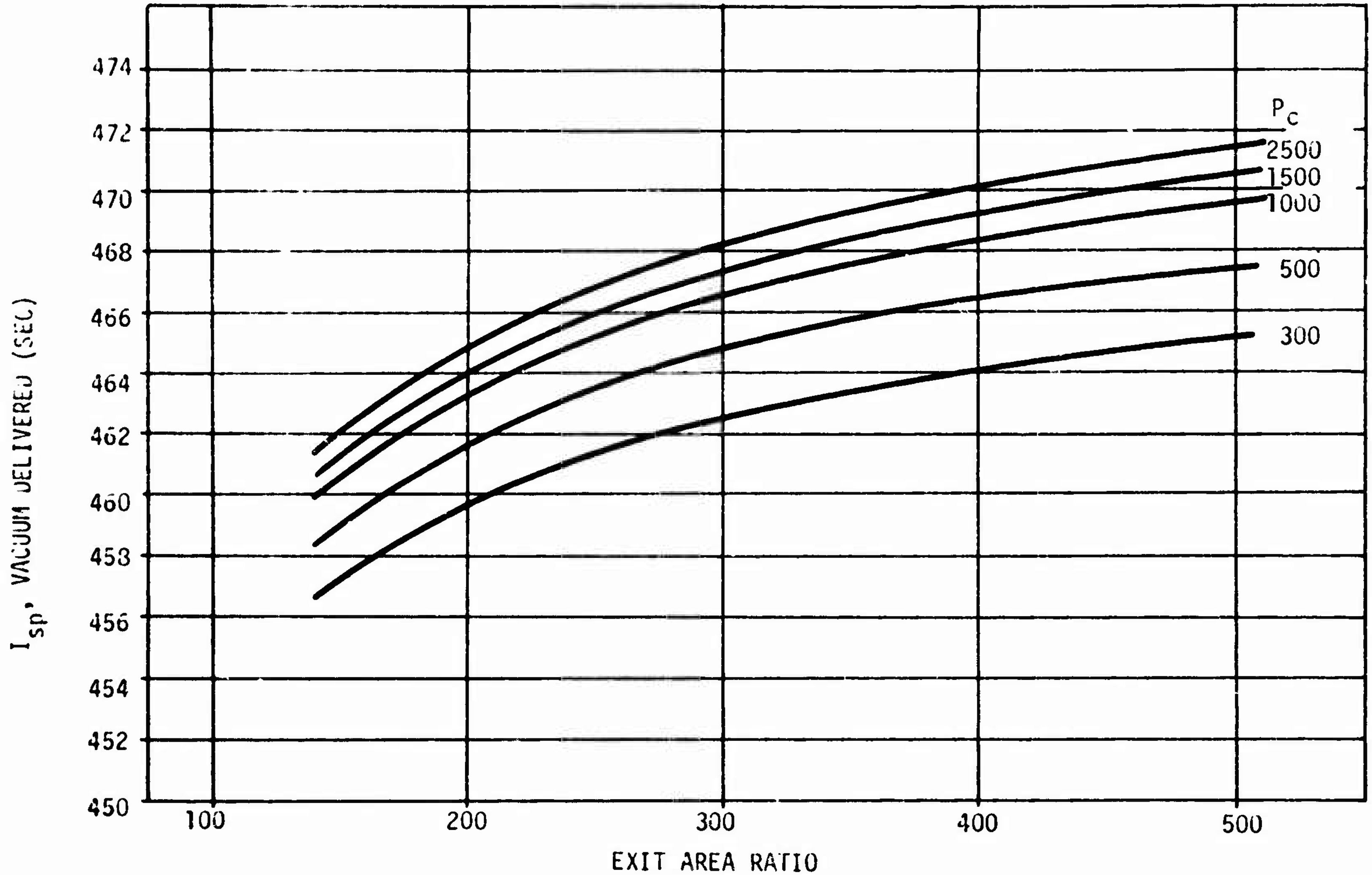


Figure 96. I_s vs Exit Area Ratio, F = 25K, MR = 5, Staged Combustion Cycle

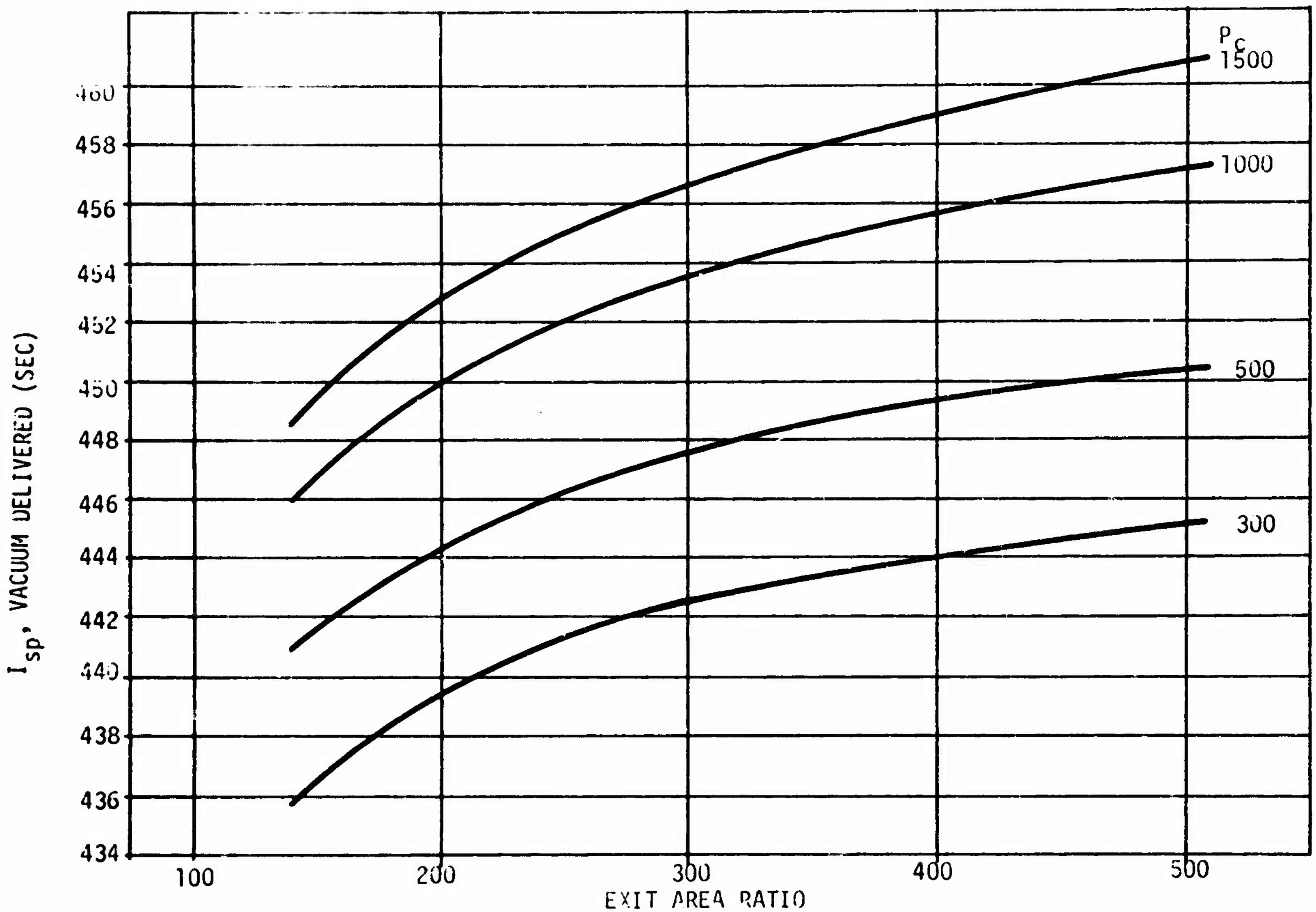


Figure 97. I_s vs Exit Area Ratio, F = 8K, MR = 7, Staged Combustion Cycle

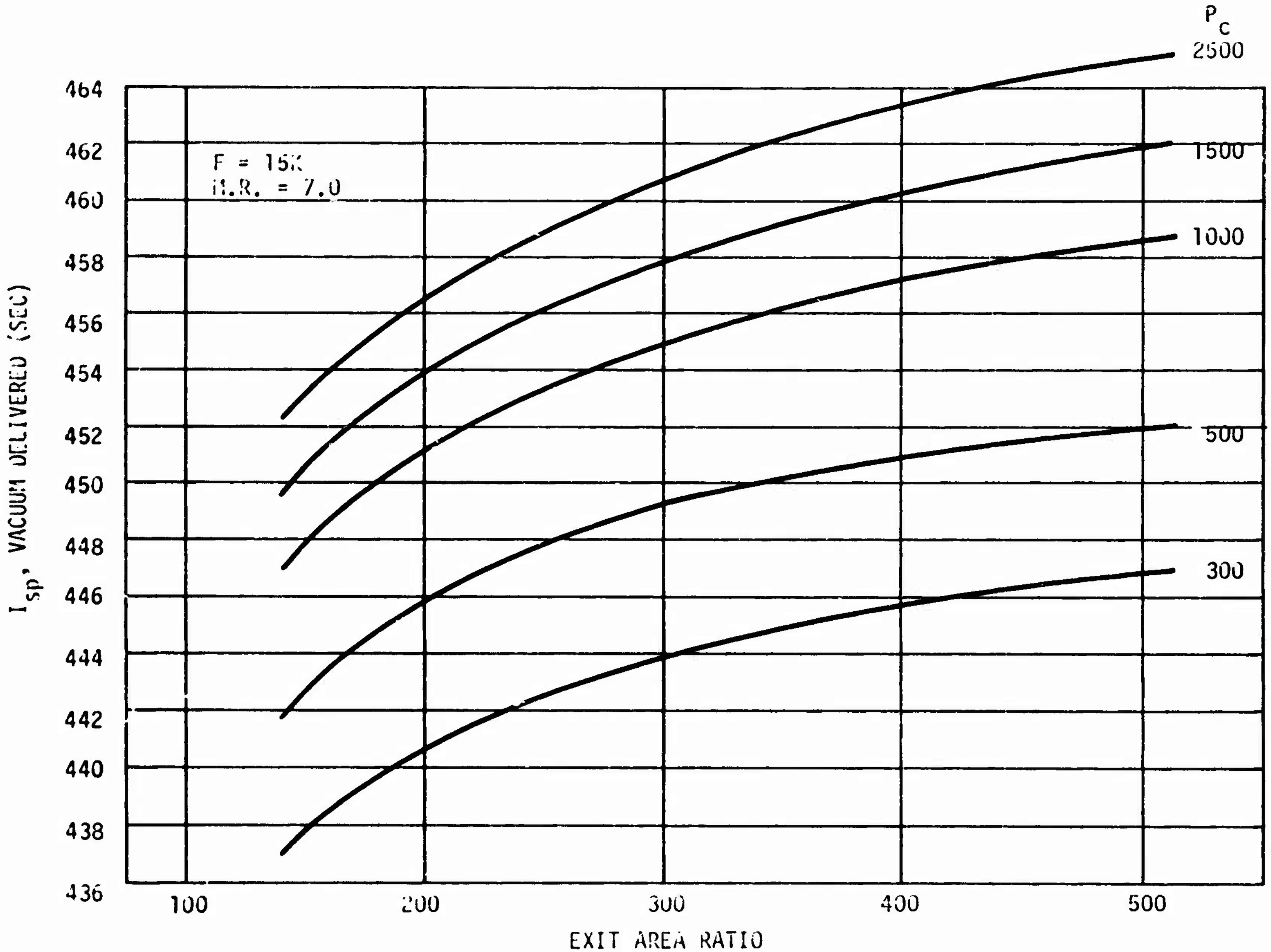


Figure 98. I_s vs Exit Area Ratio, F = 15K, MR = 7, Staged Combustion Cycle

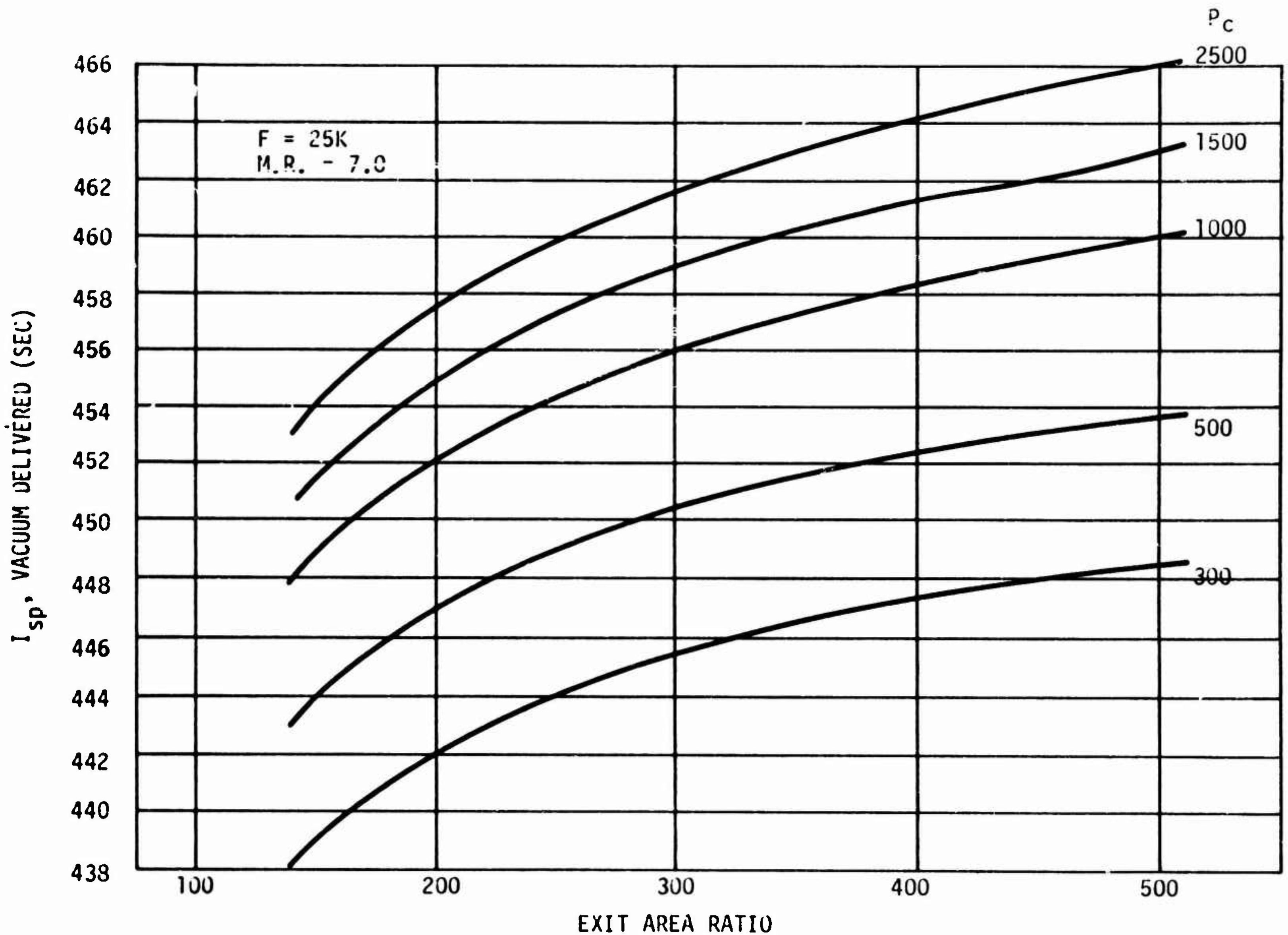


Figure 99. I_s vs Exit Area Ratio, F = 25K, MR = 7, Staged Combustion Cycle

Figures 100 through 105 indicate the calculated engine weights for the three thrust levels also as a function of chamber pressure and nozzle expansion area ratio for the fixed, minimum weight retractable and minimum length retractable nozzles.

The engine envelope effects are shown in Figures 106 and 107 for MR = 5.0 and in Figures 108 and 109 for MR = 7.0 as a function of the thrust to chamber pressure ratio, and nozzle expansion area ratio. These figures indicate changes in engine envelope due to the changing nozzle contour required for different design mixture ratios. The engine lengths are operating engine lengths meeting the stowed engine length requirements.

Additional information is shown indicating effects of envelope constraints on achievable nozzle area ratios for both the fixed and retractable nozzle concept. Figure 106 indicates the feasible expansion area ratios for an engine mixture ratio 5.0 as function of thrust to chamber pressure ratio for both nozzle concepts. Figure 107 indicates that area ratio at which the retractable nozzle has to be split to stowed the nozzle in the available stowed envelope. Identical information is provided for MR = 7.0 in Figures 108 and 109.

Figures 110 and 111 summarize the interactions at mixture ratios of 5 and 7, respectively. The effect on suction line diameters were calculated as a function of mixture ratio and discrete NPSH values. The fuel line diameter is very insensitive to the NPSH requirement. However, the oxidizer suction line is very dependent upon thrust and NPSH requirements. The results are indicated in Figure 112 and Figure 113 and are calculated with the precedures described in Section III,B,2,g.

NPSH effect on engine weight is very small since only the low speed pump weights are affected and only the oxygen low speed pump is significantly effected. The hydrogen low speed pump can take advantage of the TSH (thermal suppression head) and therefore is very insensitive. The results of the analysis are shown in Figure 114.

The staged combustion cycle characteristic is presented for the selected chamber pressures for three mixture ratios, and the NPSH values required. The characteristics are summarized in Table XVIII for both nozzle concepts where applicable.

For these selected chamber pressures, the engine flow schematic is shown for mixture ratios of 5.0 and 7.0. No influence of NPSH is shown in these schedules presented in Figures 115 and 116.

The effect of the turbomachinery design conditions are summarized in Table XIX for both the oxygen and fuel circuit.

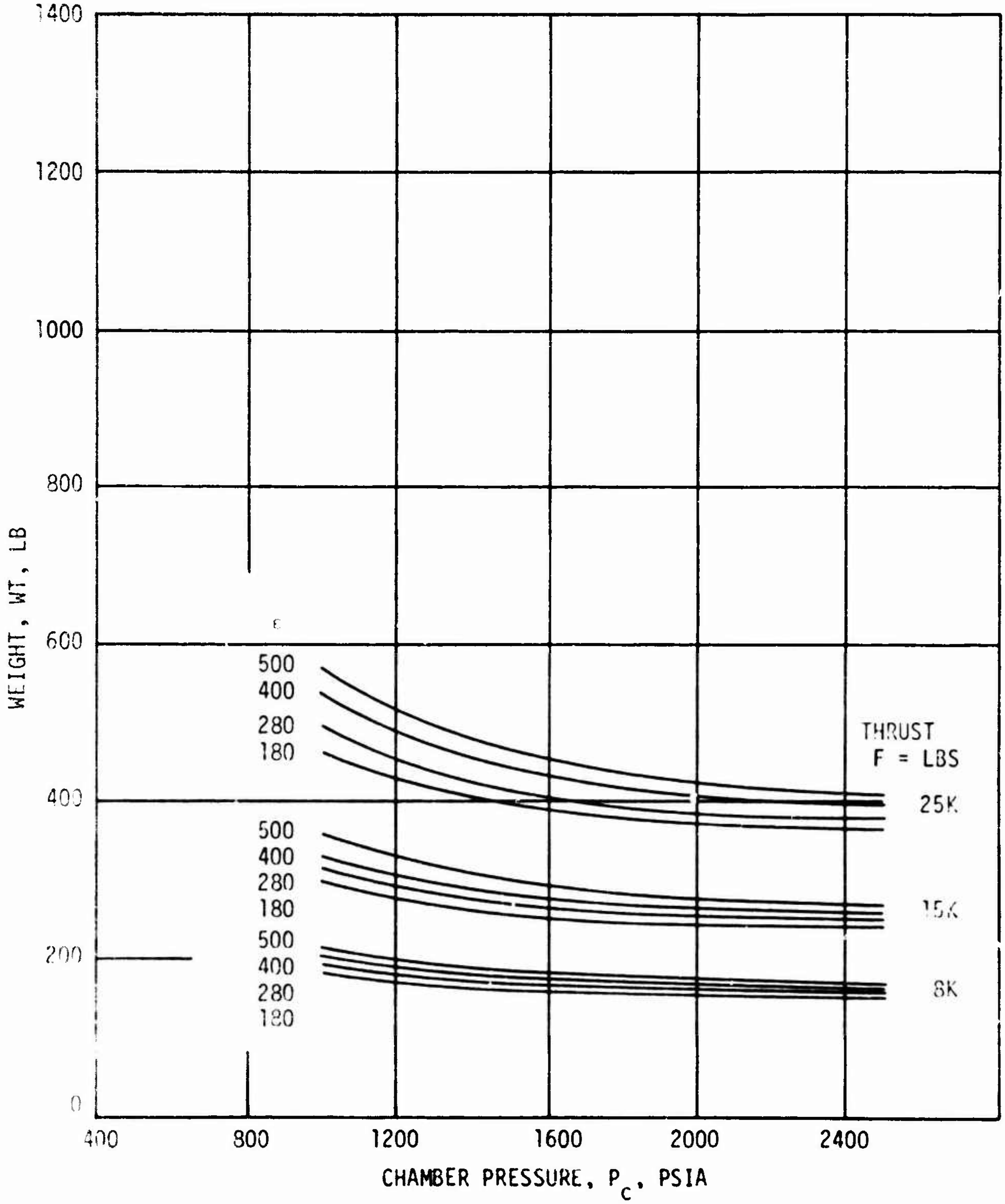


Figure 100. Weight vs P_c, Fixed Nozzle, MR = 5

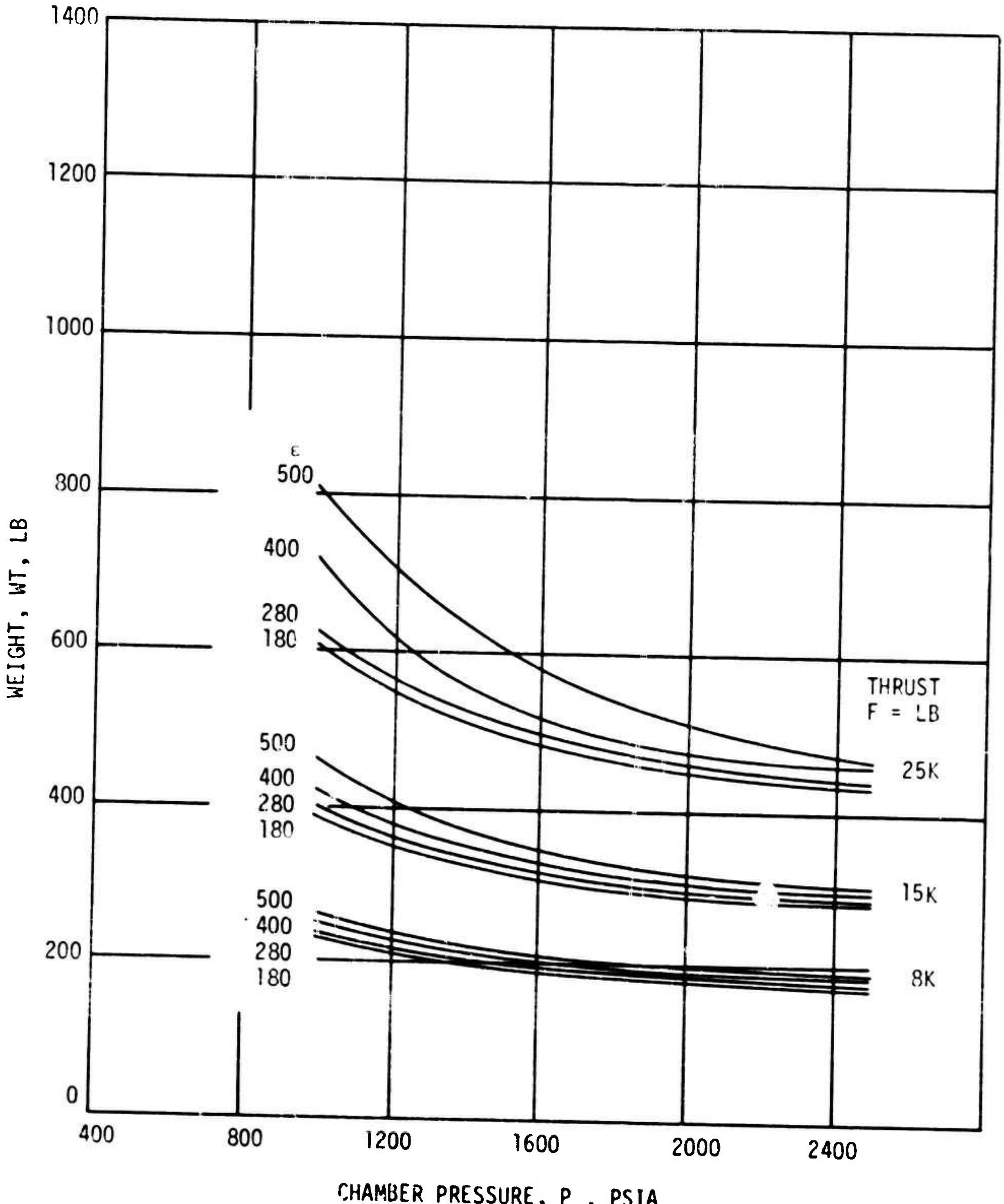


Figure 101. Weight vs P_c, Minimum Weight Retractable Nozzle, MR = 5

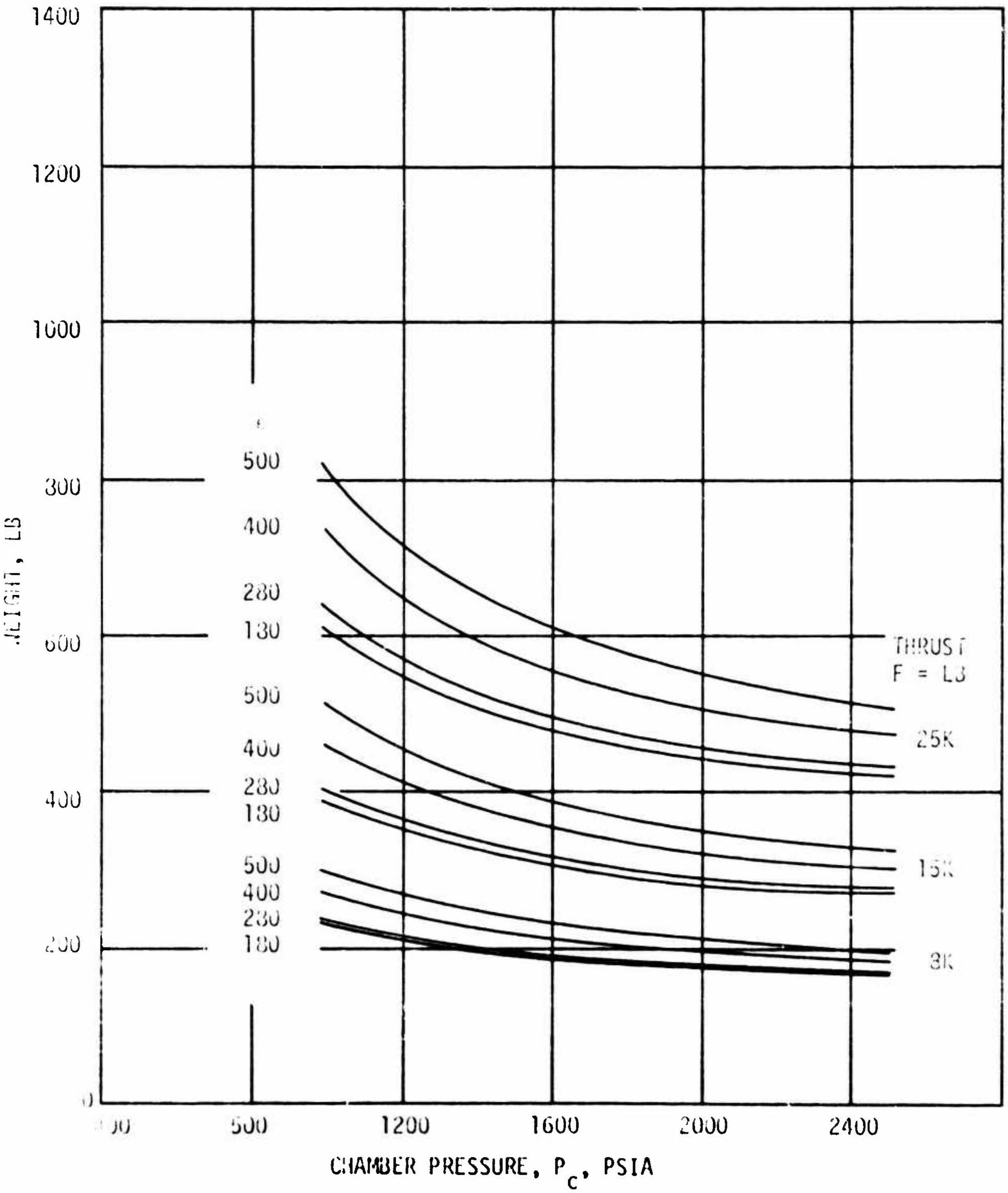


Figure 102. Weight vs P_c, Minimum Length Retractable Nozzle, MR = 5

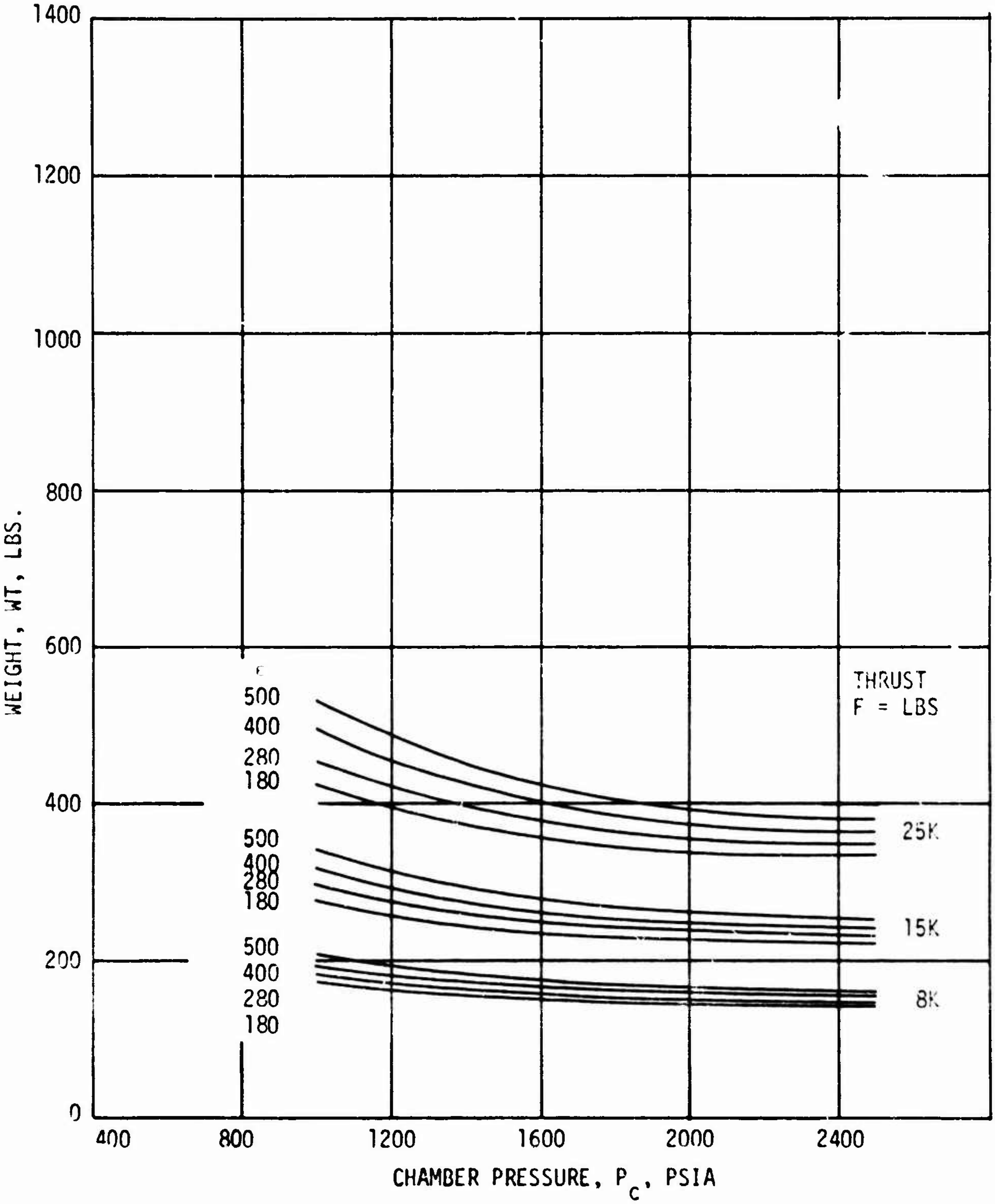


Figure 103. Weight vs P_c, Fixed Nozzle, MR = 7

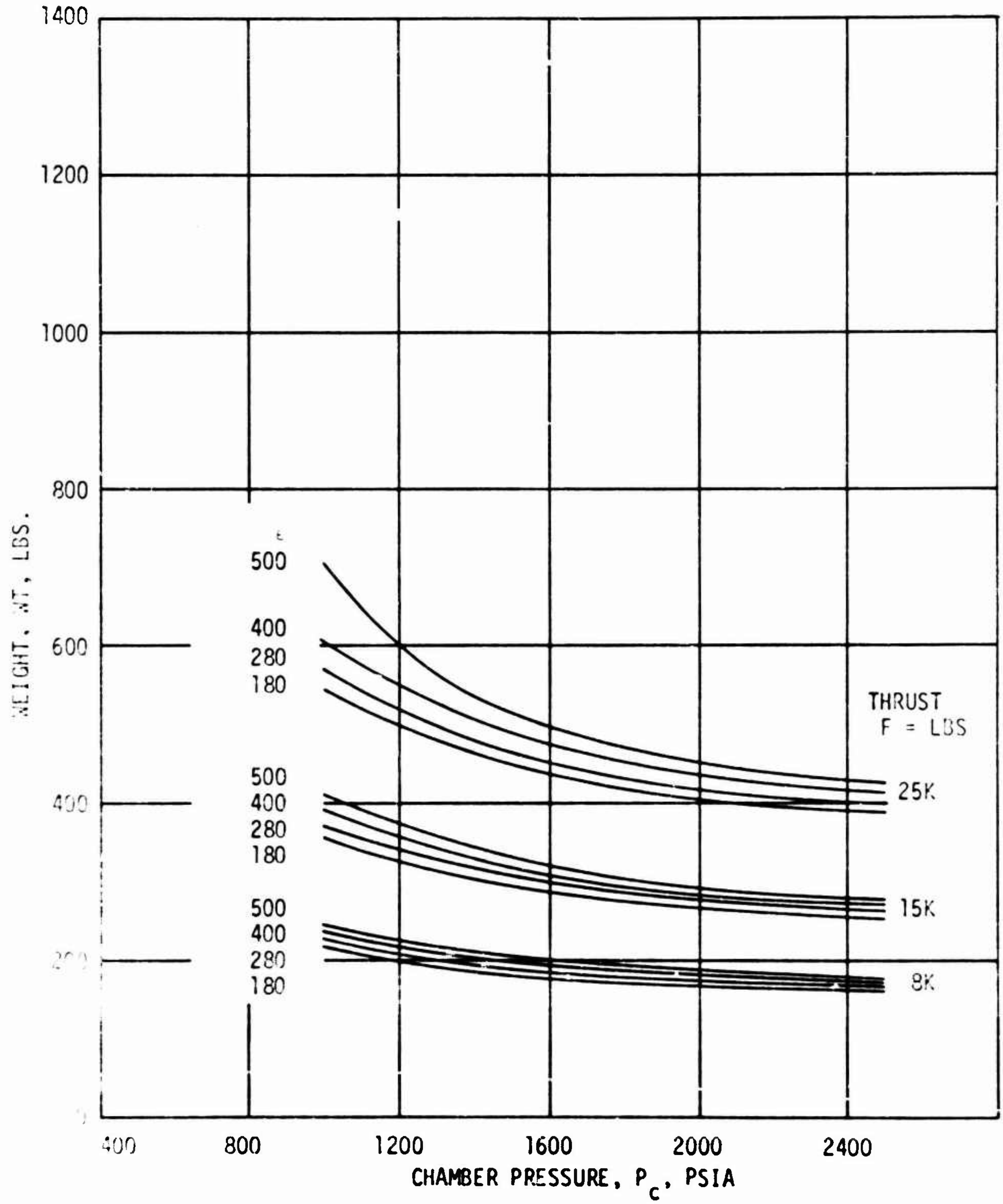


Figure 104. Weight vs P_c, Minimum Weight Retractable Nozzle, MR = 7

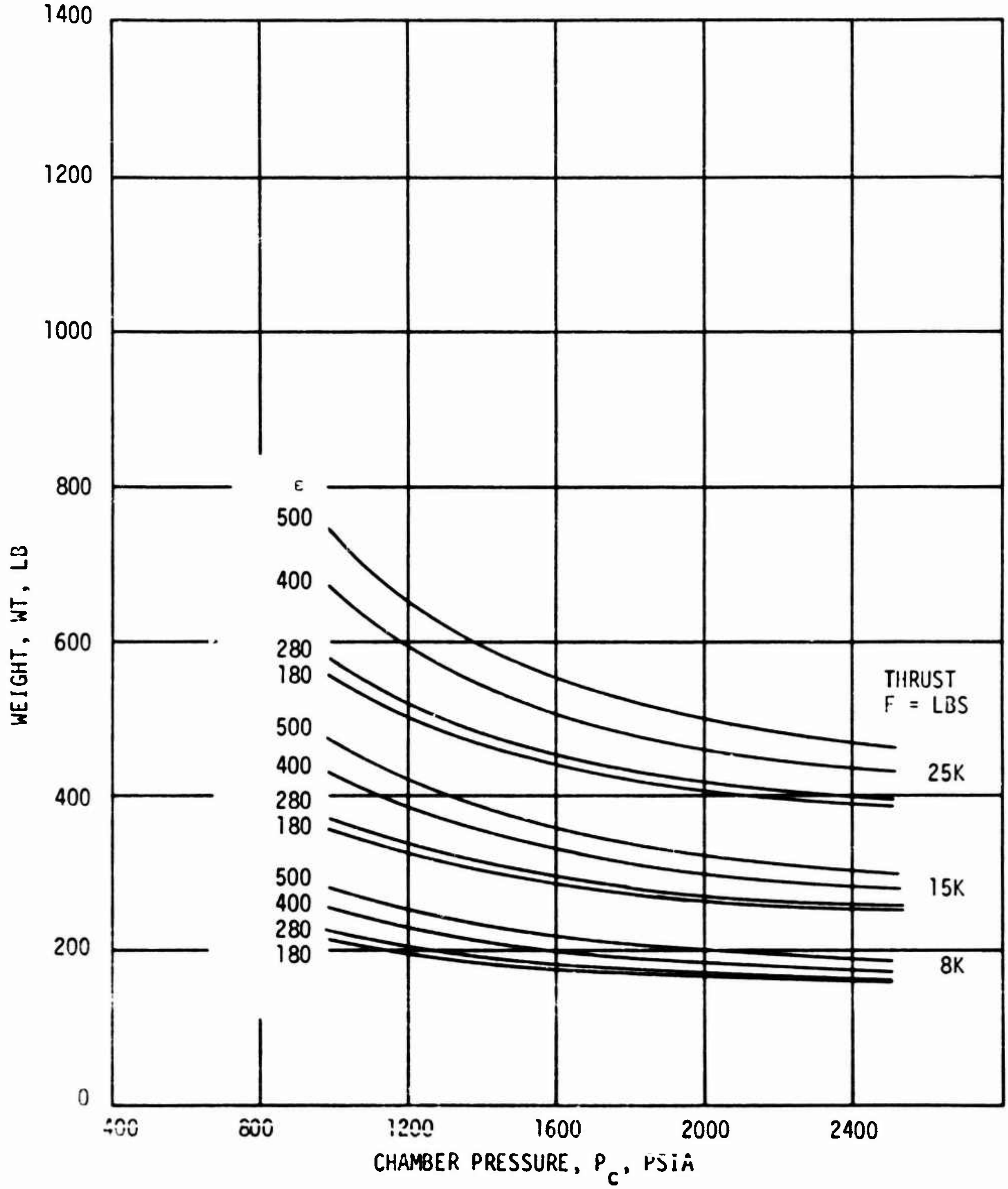


Figure 105. Weight vs P_c, Minimum Length Retractable Nozzle, MR = 7

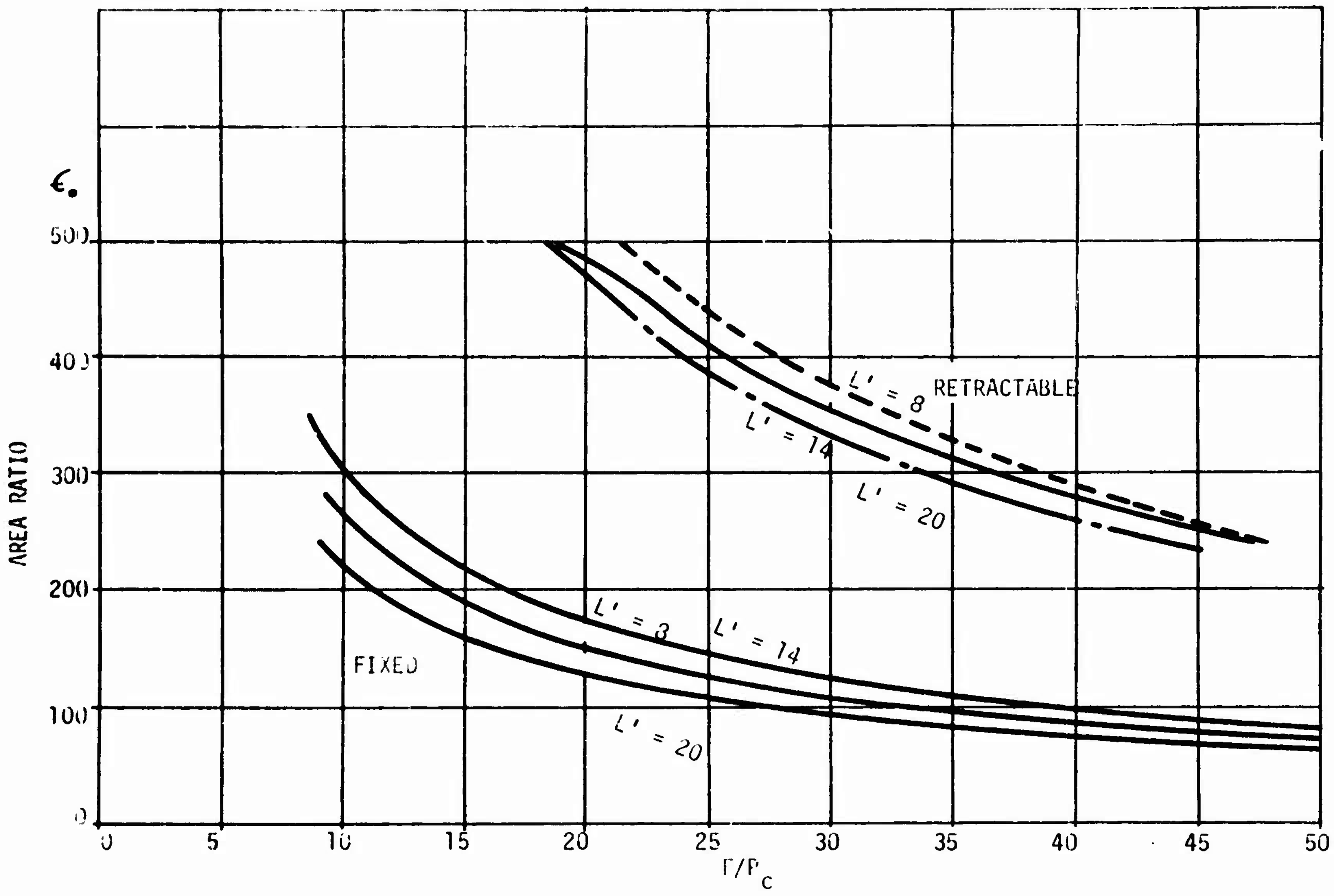


Figure 106. Maximum Area Ratio for Fixed Nozzles, Minimum Weight Retractable Nozzles

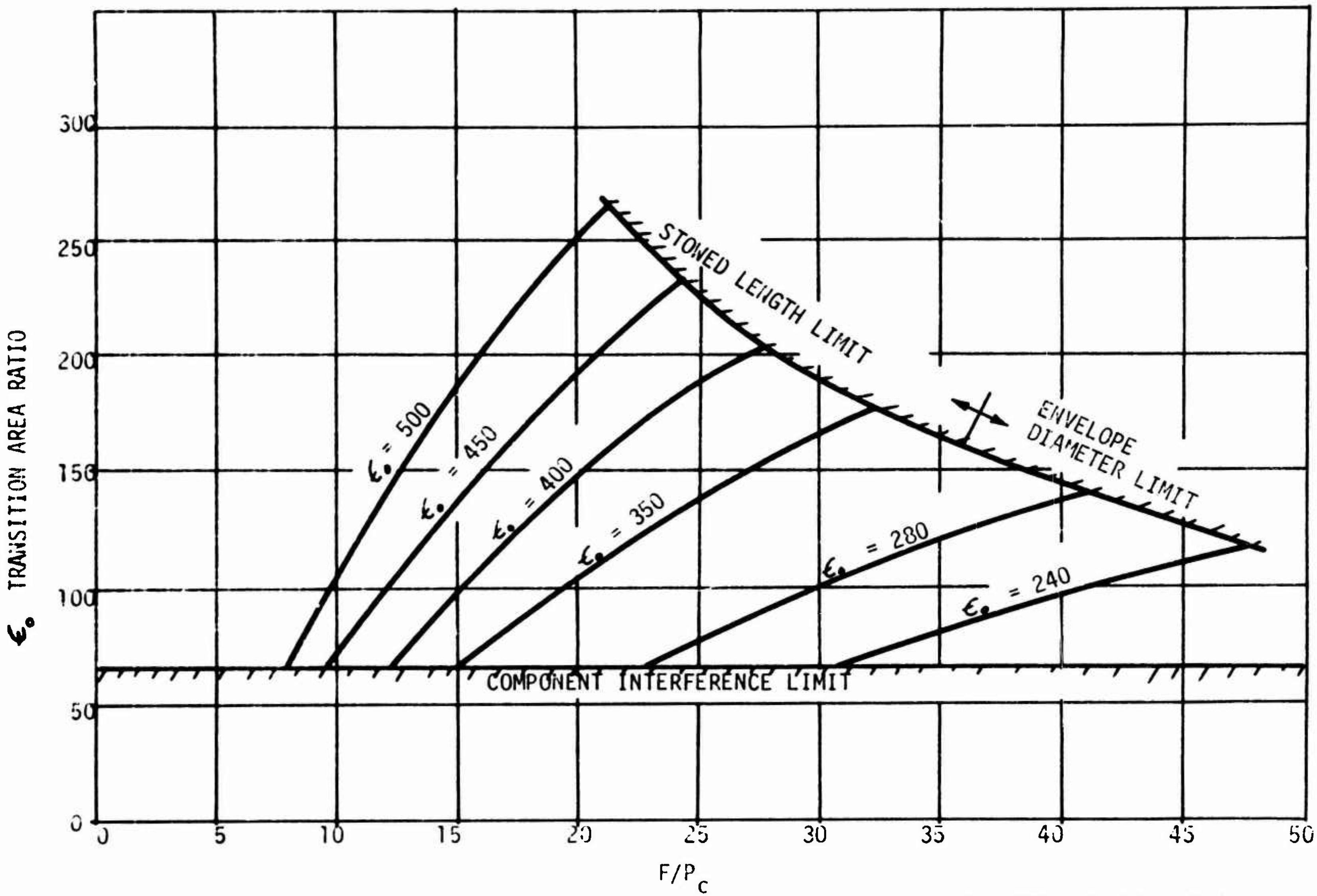


Figure 107. Transition Area Ratio for Minimum Weight Nozzle, MR = 5, L' = 8-in.

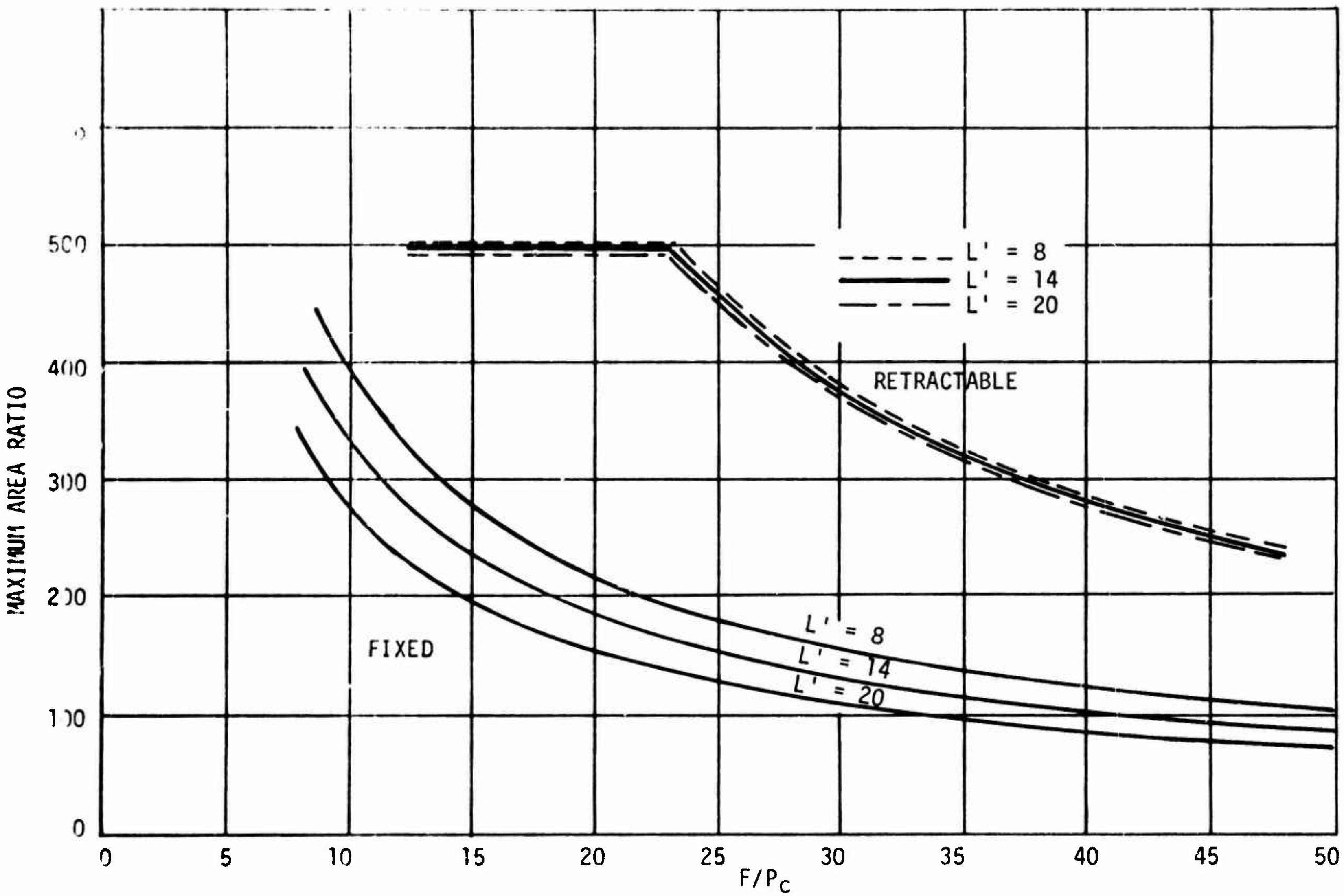


Figure 108. Maximum Area Ratio for Fixed Nozzles and Minimum Weight Retractable Nozzles, MR = 7

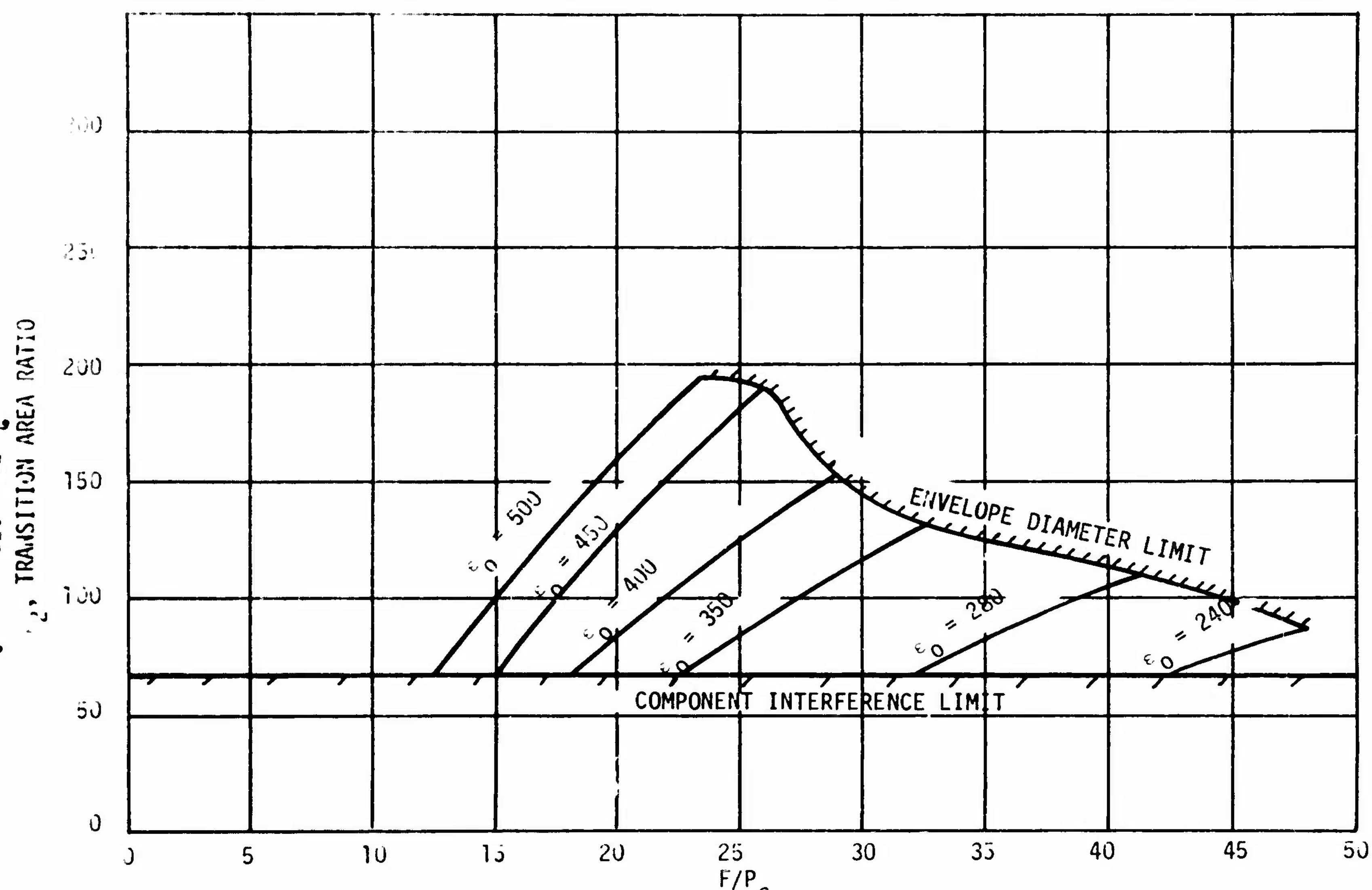


Figure 109. Transition Area Ratio for Minimum Weight Nozzle, MR = 7, L' = 8-in.

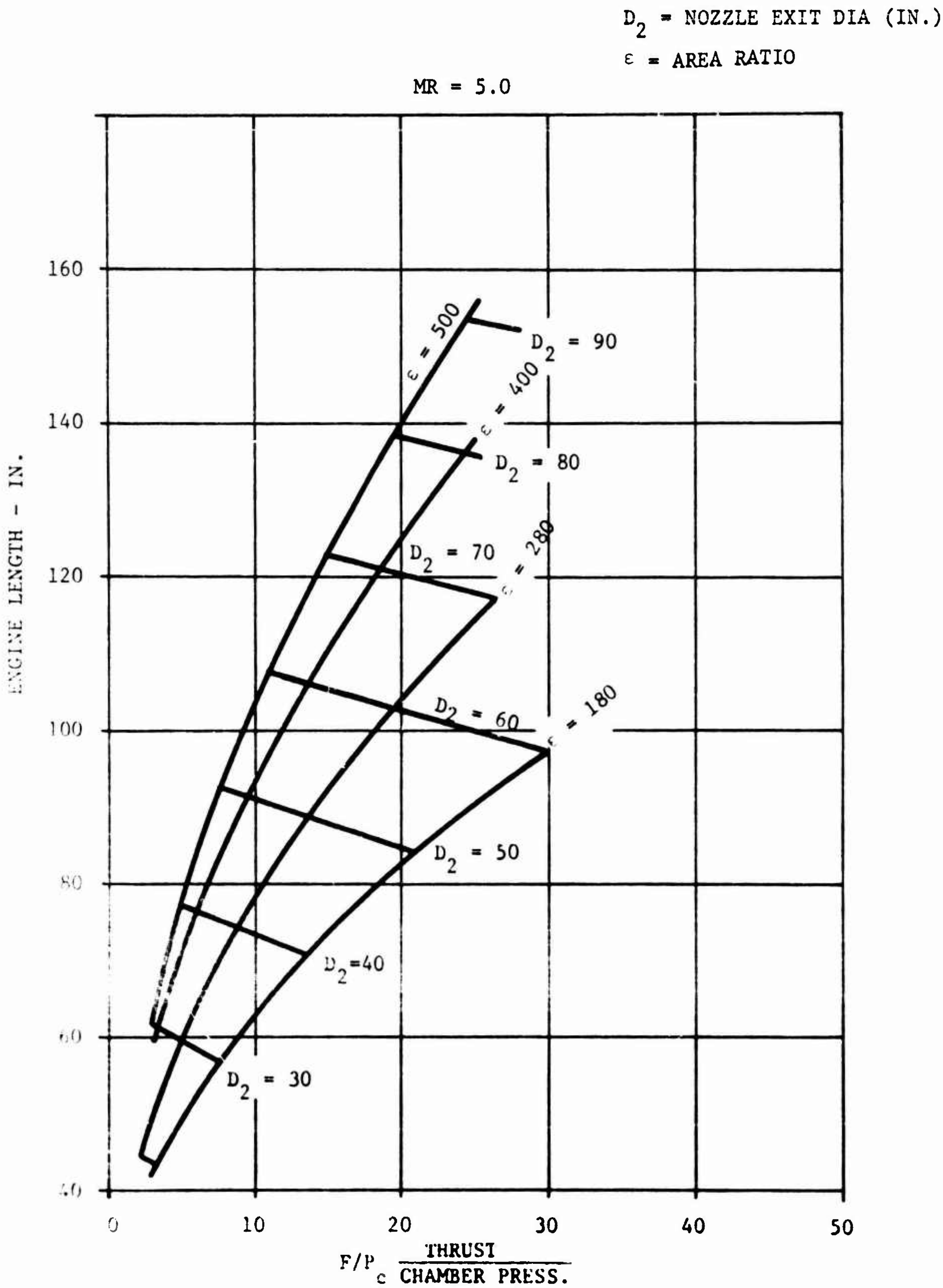


Figure 110. Engine Length vs F/P_c, MR = 5, Staged Combustion Cycle

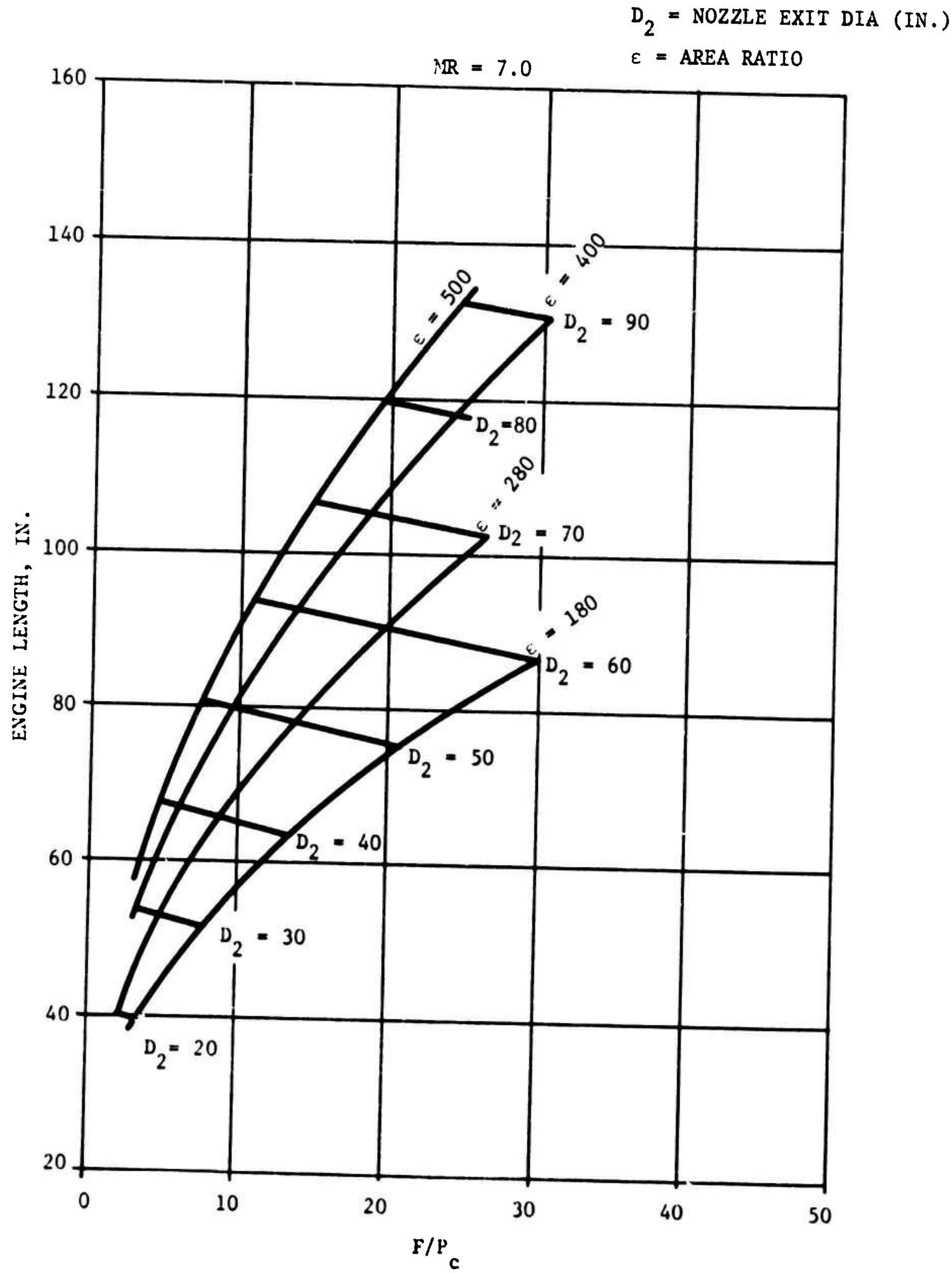


Figure 111. Engine Length vs F/P_c, MR = 7, Staged Combustion Cycle

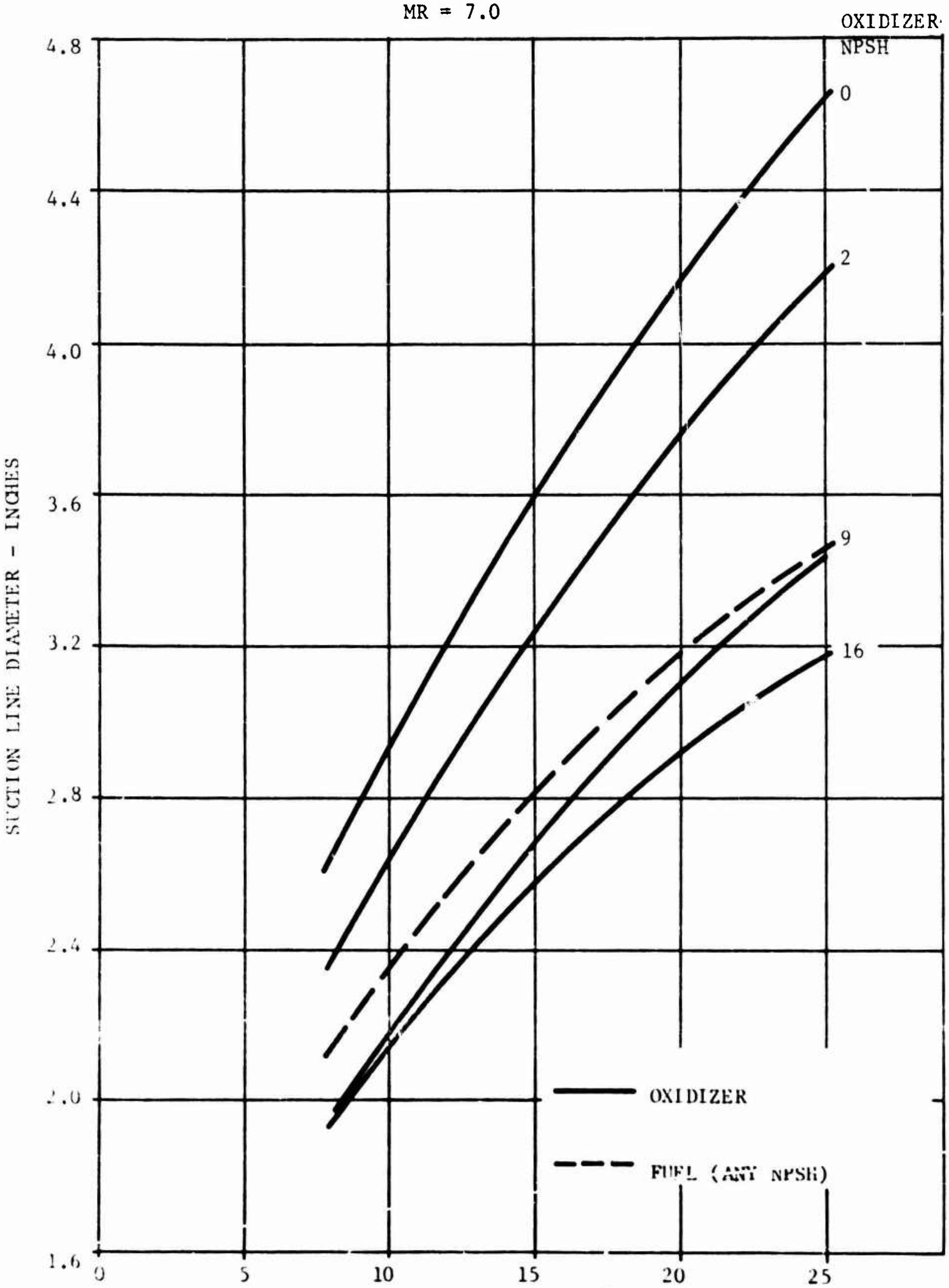


Figure 112. Suction Line Diameter vs Thrust, MR = 7

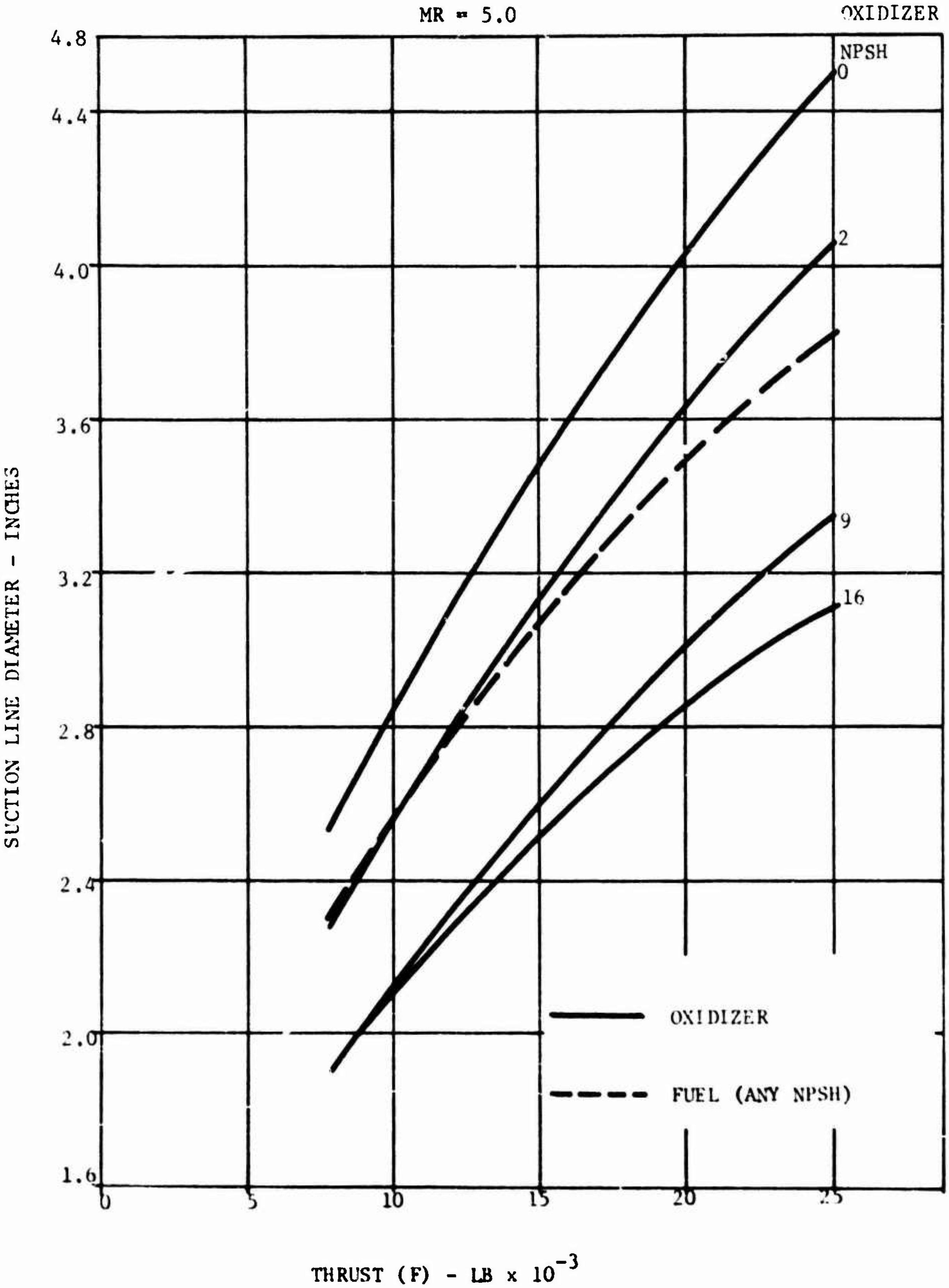


Figure 113. Suction Line Diameter vs Thrust, MR = 5

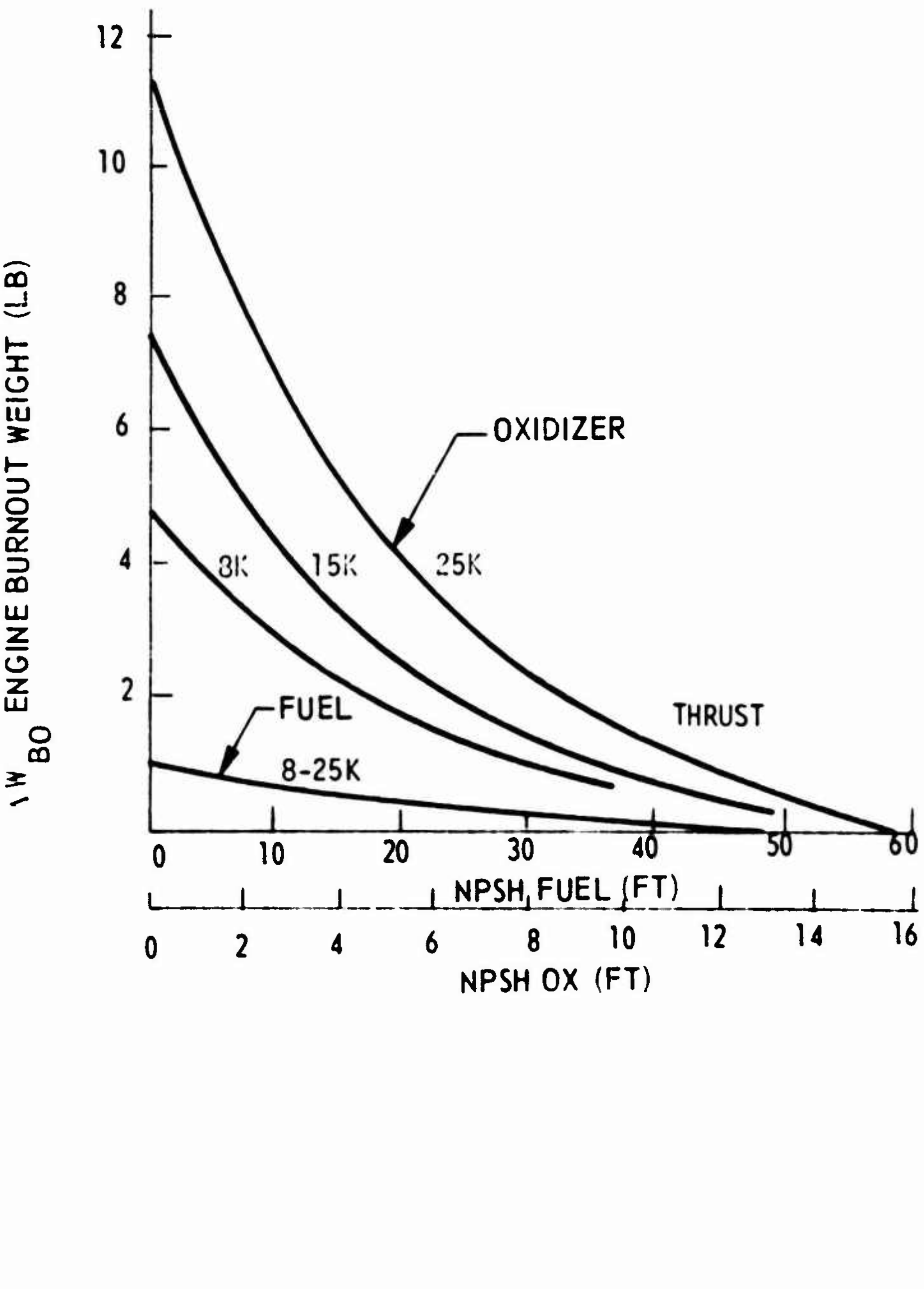


Figure 114. Staged Combustion Cycle Engine Weight vs NPSH, MR = 6

TABLE XVIII

STAGED COMBUSTION CYCLE CHARACTERISTICS FOR VARIOUS P_c, MR AND NPSH

Nozzle Concept		Fixed		
Thrust	1000 lb	8	8	8
Mixture Ratio		5	6	7
Chamber Pressure	psia	1200	1200	1200
Area Ratio		450	500	500
Specific Impulse	sec	469.4	466.5	458.5
Engine Length	in.	82	82	78.0
Maximum Diameter	in.	42.5	47.5	47.5
Suction Line Orientation	in.	15	15	15
NPSH = 15/2 F/O	Engine Weight lb	200.5	200.5	197.5
	Suction Line dia in. Fuel	2.33	2.20	2.14
	Ox	2.30	2.33	2.38
NPSH = 27/9 F/O	Engine Weight lb	199	199	197
	Suction Line dia in. Fuel	2.33	2.20	2.14
	Ox	1.90	1.90	1.95
NPSH = 60/16 F/O	Engine Weight lb	198	198	196
	Suction Line dia in. Fuel	2.33	2.20	2.14
	Ox	1.90	1.90	1.95

TABLE XVIII (cont.)

Nozzle Concept		Fixed			Retractable		
Thrust	1000 lb	15	15	15	15	15	15
Mixture Ratio		5	6	7	5	6	7
Chamber Pressure	psia	1450	1450	1450	1450	1450	1450
Area Ratio		295	350	390	500	500	500
Specific Impulse	sec	467.0	466.0	460.20			
Engine Length	in.	82	82	82	106	100	93
Maximum Diameter	in.	45	49	51	59	61	60
Suction Line Orientation	in.	18	18	13	18	18	18
NPSH = 15/2 F/O	Engine Weight lb	271	275	291	365	345	335
	Suction Line dia in. Fuel	3.06	2.90	2.80	3.06	2.90	2.80
	Ox	3.13	3.18	3.24	3.13	3.18	3.24
NPSH = 27/9 F/O	Engine Weight lb	267.5	271.5	287.5	361.5	341.5	330
	Suction Line dia in. Fuel	3.60	2.90	2.80	3.60	2.90	2.80
	Ox	2.60	2.60	2.67	2.60	2.60	2.60
NPSH = 60/16 F/O	Engine Weight lb	266	270	286	360	340	330
	Suction Line dia in. Fuel	3.06	2.90	2.80	3.06	2.90	2.80
	Ox	2.60	2.60	2.67	2.60	2.60	2.67

TABLE XVIII (cont.)

Nozzle Concept			Fixed			Retractable		
Thrust		1000 lb	25	25	25	25	25	25
Mixture Ratio			5	6	7	5	6	7
Chamber Pressure		psia	1800	1800	1800	1800	1800	1800
Area Ratio			260	280	315	500	500	500
Specific Impulse		sec	466.2	465.3	460.2	471.0	469.8	463.7
Engine Length		in.	82	82	82	120	110	104
Maximum Diameter		in.	46.3	50.5	53.0	68	68	68
Suction Line Orientation		in.	24	24	24	24	24	24
NPSH = 15/2	Engine Weight lb		394	389	379	549	599	484
	Suction Line dia in.	Fuel	3.80	3.60	3.46	3.80	3.60	3.46
		Ox	4.05	4.10	4.18	4.05	4.10	4.18
NPSH = 27/9	Engine Weight lb		387	382	372	542	492	477
	Suction Line dia in.	Fuel	3.80	3.60	3.46	3.80	3.60	3.46
		Ox	3.35	3.40	3.44	3.35	3.40	3.44
NPSH = 60/16	Engine Weight lb		385	380	370	540	490	475
	Suction Line dia in.	Fuel	3.80	3.60	3.46	3.80	3.60	3.46
		Ox	2.90	3.12	3.17	2.90	3.12	3.17

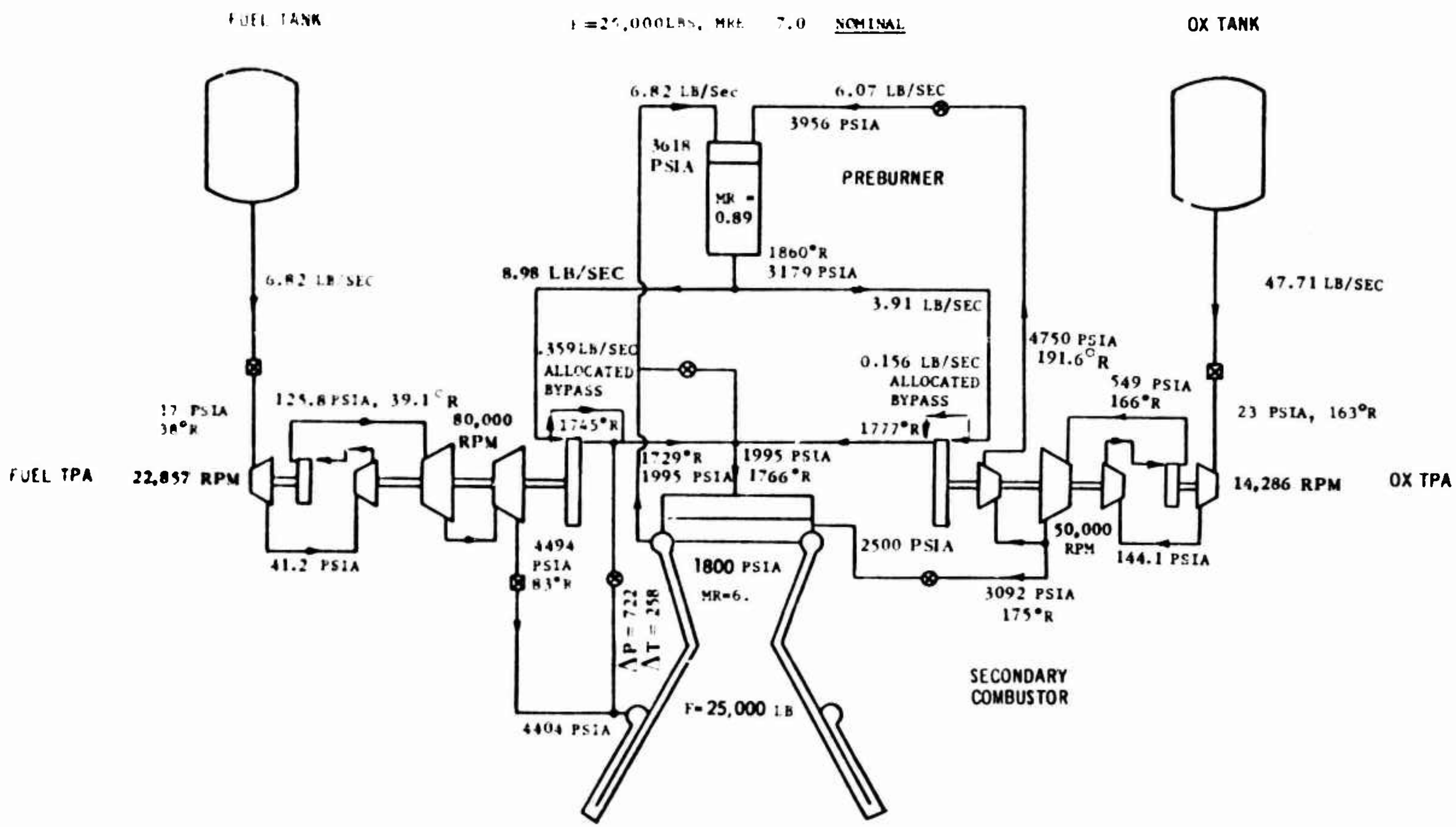


Figure 115. Engine Schematic, Staged Combustion Cycle, MR = 7

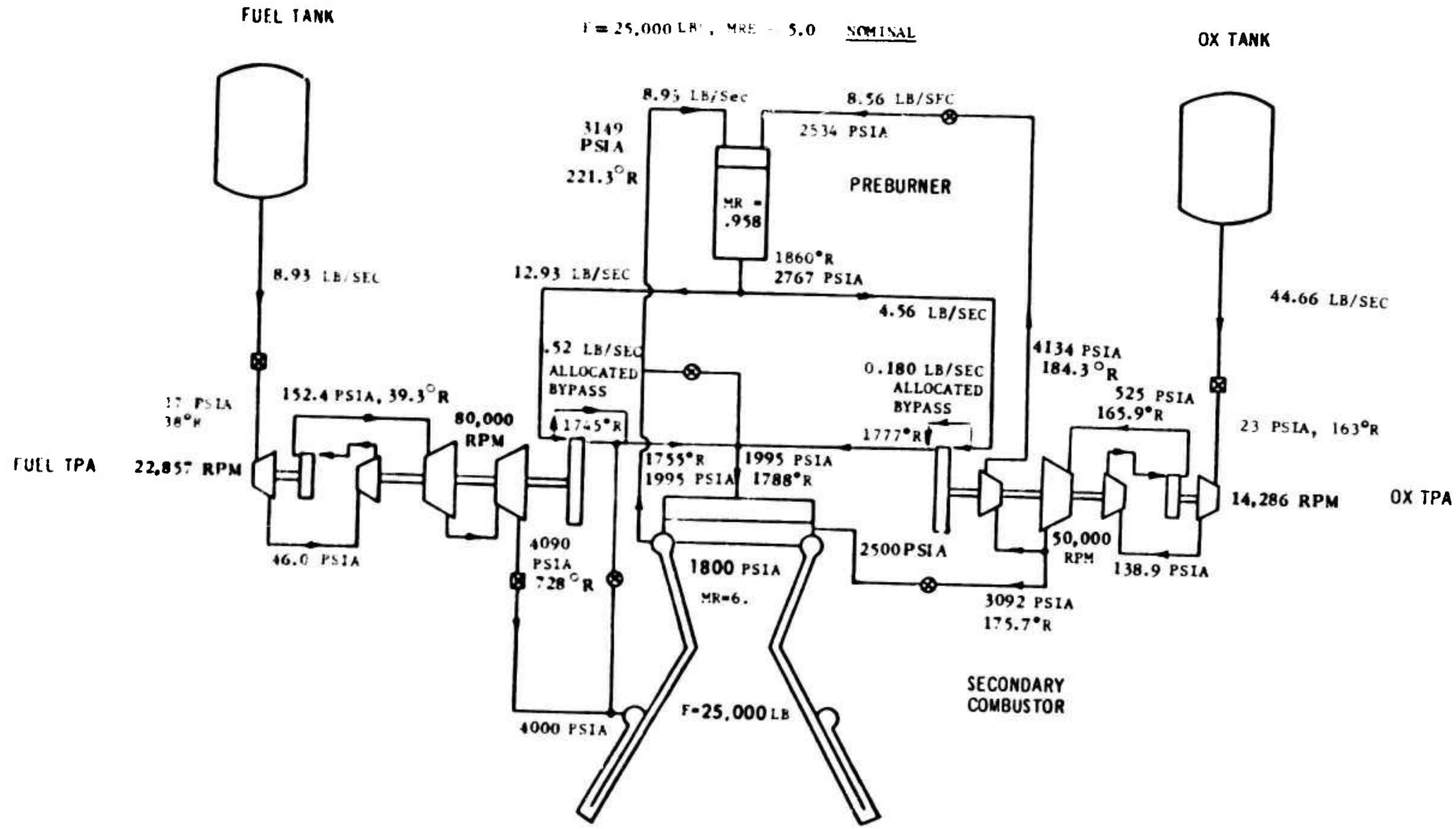


Figure 116. Engine Schematic, Staged Combustion Cycle, MR = 5

TABLE XIX

EFFECT OF TURBOMACHINERY DESIGN

Engine MR = 7   NPSH = 60 ft

Fuel Circuit		Thrust 8K	15K	25K
Boost Pump				
Shaft Speed	rpm	31428	25714	22857
Suction Specific Speed	-	9232	10348	11884
Main Pump				
Number Stages	-	2	2	3
Shaft Speed	rpm	11000	90000	80000
Impeller Tip Speed	ft/sec	1787	1726	1515
Specific Speed	-	462	465	652
Efficiency	%	49.8	52.6	59.7
Discharge Pressure	psia	3388	3907	4494
Volume Flow	gpm	225	421	704
Turbine				
Inlet Temperature	°R	1860	1860	1860
Mean Blade Speed	ft/sec	1300	1300	1300
Number Stages	-	2	2	2
Nozzle Admission	%	100	100	100
Efficiency	%	64.1	62.7	62.1
Shaft Power	HP	929	1904	3225

TABLE XIX (cont.)

Engine MR = 7   NPSH = 16 ft

Oxidizer Circuit		Thrust 8K	15K	25K
Boost Pump				
Shaft Speed	rpm	20000	16000	14286
Suction Specific Speed	-	20727	22717	26207
Main Pump				
Number Stages	-	1-1/2	1-1/2	1-1/2
Shaft Speed	rpm	70000	56000	50000
Impeller Tip Speed	ft/sec	490	534	593
Specific Speed	-	1552	1458	1428
Efficiency	%	57.9	61.4	64.1
Discharge Pressure*	psia	2062	2491	3092
Volume Flow	gpm	96.1	180	301
Turbine				
Inlet Temperature	°R	1860	1860	1860
Mean Blade Speed	ft/sec	1100	1100	1100
Number Stages	-	1	1	1
Nozzle Admission	%	24	23.7	27.7
Efficiency	%	46.6	45.3	44.7
Shaft Power	HP	235	506	1012

* Main Stage

TABLE XIX (cont.)

Engine MR = 5 NPSH = 60 ft

Fuel Circuit		Thrust 8K	15K	25K
Boost Pump				
Shaft Speed	rpm	31428	25714	22857
Suction Specific Speed	-	10584	11855	13605
Main Pump				
Number Stages	-	2	2	3
Shaft Speed	rpm	11000	90000	80000
Impeller Tip Speed	ft/sec	1549	1640	1503
Specific Speed	-	559	575	807
Efficiency	%	53.9	57.0	63.6
Discharge Pressure	psia	3170	3559	4090
Volume Flow	gpm	295	553	922
Turbine				
Inlet Temperature	°R	1860	1860	1860
Mean Blade Speed	ft/sec	1300	1300	1300
Number Stages	-	2	2	2
Nozzle Admission	%	100	100	100
Efficiency	%	71.1	70.6	70.3
Shaft Power	HP	1056	2102	3614

TABLE XIX (cont.)

Engine MR = 5   NPSH = 16 ft

Oxidizer Circuit		Thrust 8K	15K	25K
Boost Pump				
Shaft Speed	rpm	20000	16000	14286
Suction Specific Speed	-	20084	21997	25356
Main Pump				
Number Stages	-	1-1/2	1-1/2	1-1/2
Shaft Speed	rpm	70000	56000	50000
Impeller Tip Speed	ft/sec	487	531	590
Specific Speed	-	1493	1403	1372
Efficiency	%	57.3	60.8	63.5
Discharge Pressure*	psia	2062	2491	3092
Volume Flow	gpm	90.2	169	282
Turbine				
Inlet Temperature	°R	1860	1860	1860
Mean Blade Speed	ft/sec	1100	1100	1100
Number Stages	-	1	1	1
Nozzle Admission	%	33	34	38.8
Efficiency	%	53.7	53.0	52.0
Shaft Power	HP	223	476	947

* Main Stage

11. Thermal Conditioning Requirements

a. Engine Chilldown Analysis

An engine chilldown study was made to estimate pump chilldown requirements. The results are preliminary; however, they are sufficiently accurate to indicate the functional relationships. They should be scaled to actual emperical data and repeated for improved controlled chilldown methods. In particular, improvements are required in the LO_2 pump chilldown method.

The basic assumption of this analysis is that the pump discharge port is choked controlling the two phase flow. Obviously, improvements can be made by incorporating valve controlled chilldown particularly toward the final stage of the chilldown transient.

The parametric study was performed to determine the effect of thrust and chamber pressure on propellant pump chilldown time and amount of propellant flowed. The effect of pump inlet pressure and pump initial temperature was also studied. The range of variables for which the study was made is:

Thrust:	8000 - 50,000 lb
Chamber Pressure:	1000 - 4000 psia
Pump Inlet Pressure:	15 - 100 psia
Initial Pump Temperature:	T_{sat} - 540°R

The pumps were assumed to be made of aluminum (Table XV). The results, presented in Figures 117 through 123 are intended to be relative and may be scaled directly to match available data.

(1) Method of Analysis

The analysis was performed to provide a simple method of calculation for use in the chilldown analysis of the OOS engine. Attention was given to determination of functional relationships so that the final results could be scaled directly to available data.

The pump is modeled thermally as a one-dimensional plate, insulated on one side, with wetted area and mass equal to that of the pump. If chilldown is defined to be accomplished when a given temperature is reached then the time may be determined as a function of pump flow passage geometry and material properties.

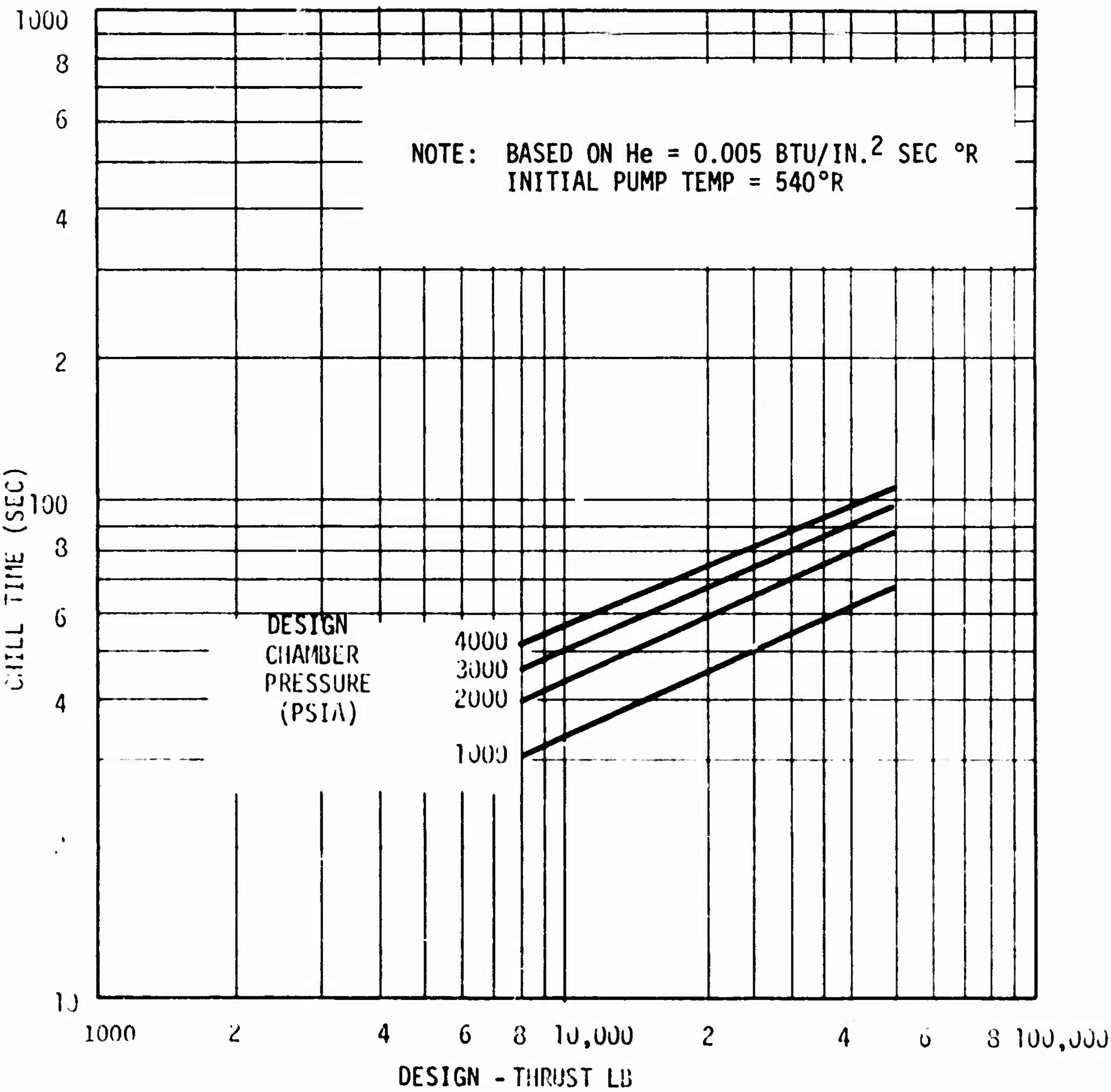


Figure 117. Hydrogen Pump Chilldown Time

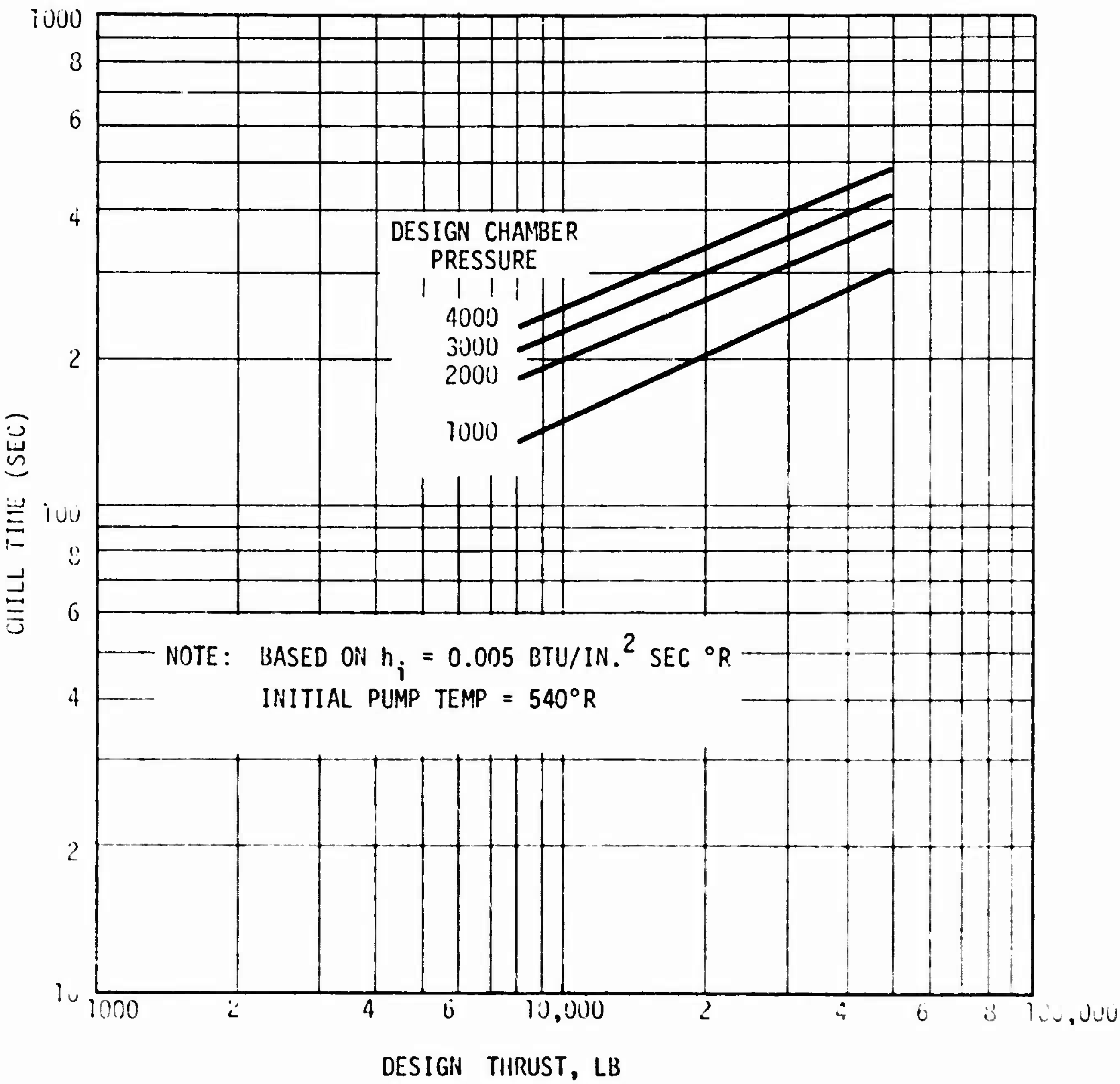


Figure 118. Oxygen Pump Childown Time

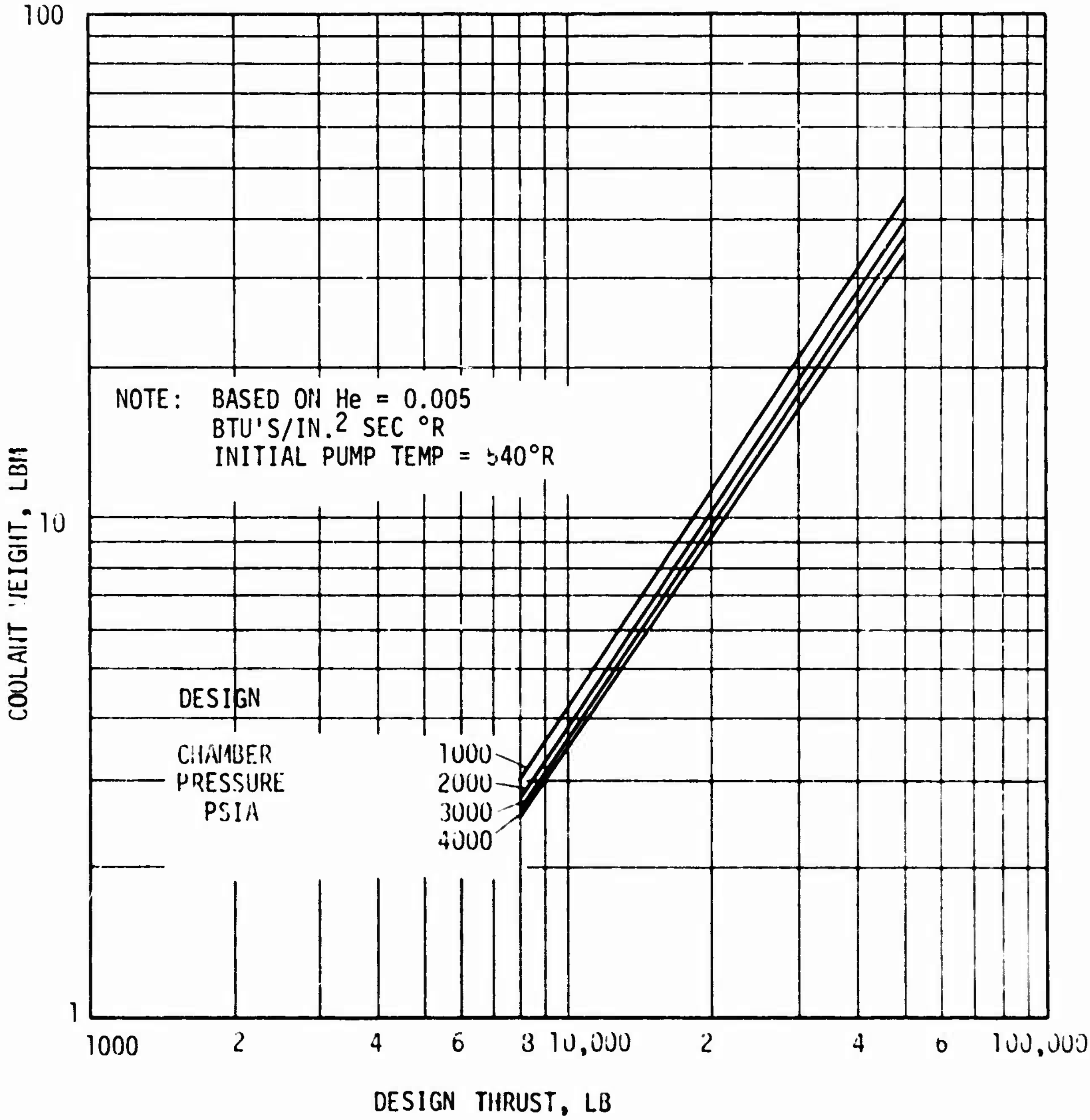
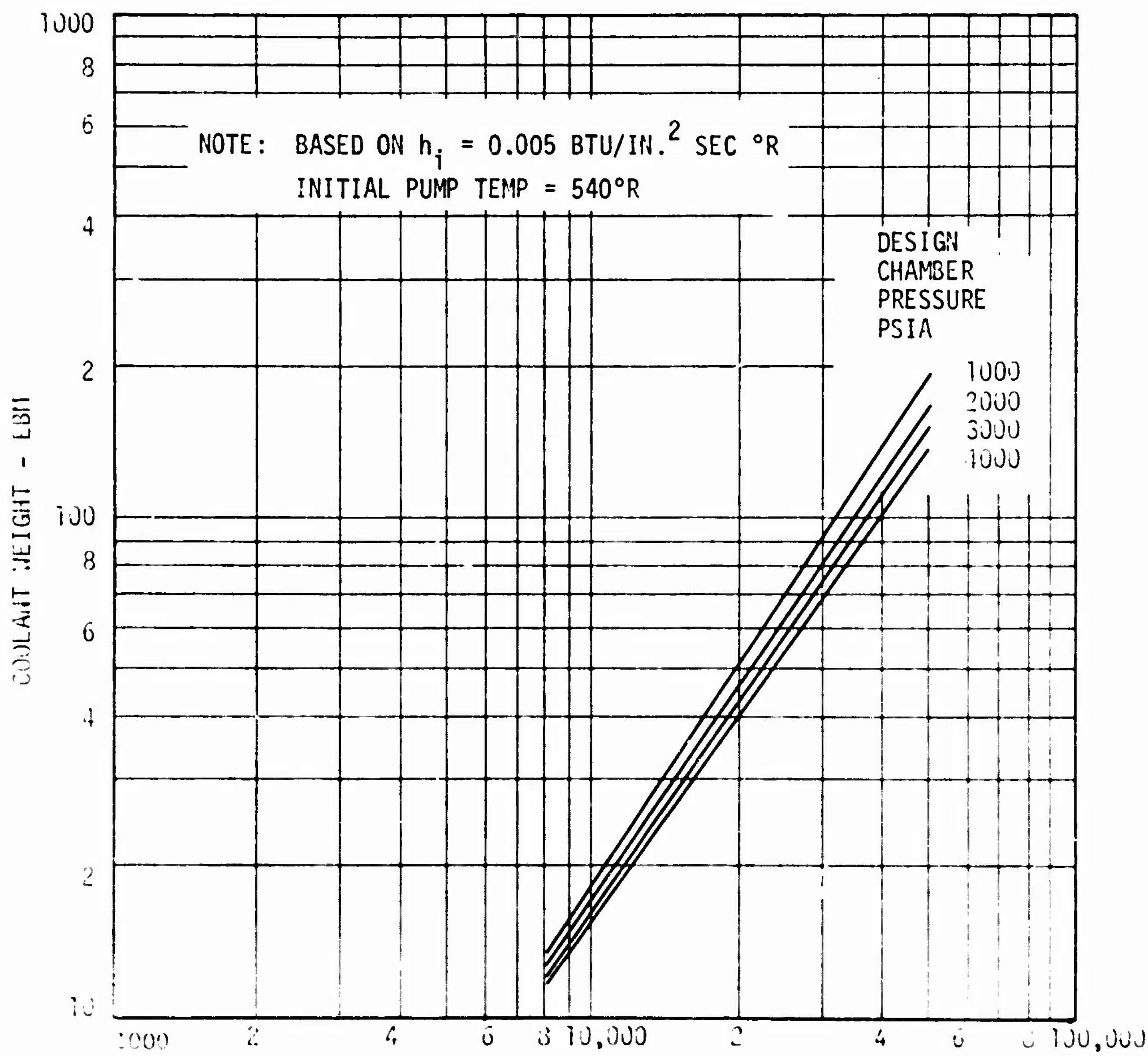


Figure 119. Hydrogen Pump Chilldown Flow



DESIGN THRUST, LB

Figure 120. Oxygen Pump Childown Flow

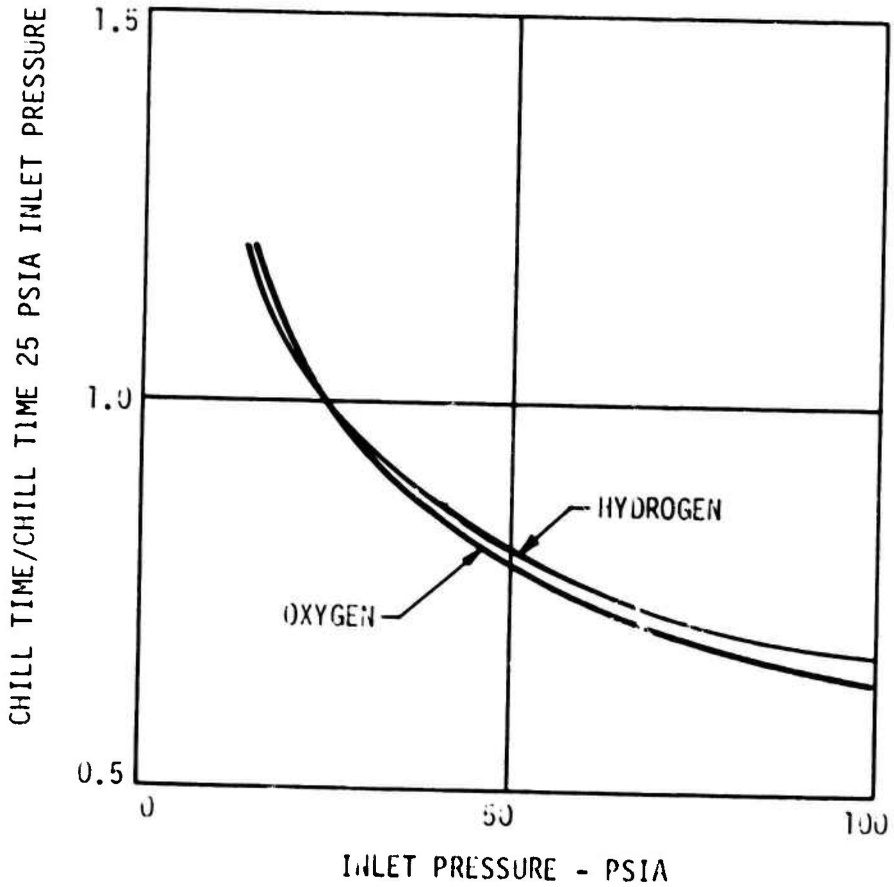


Figure 121. Effect of Inlet Pressure on Pump Childown Time

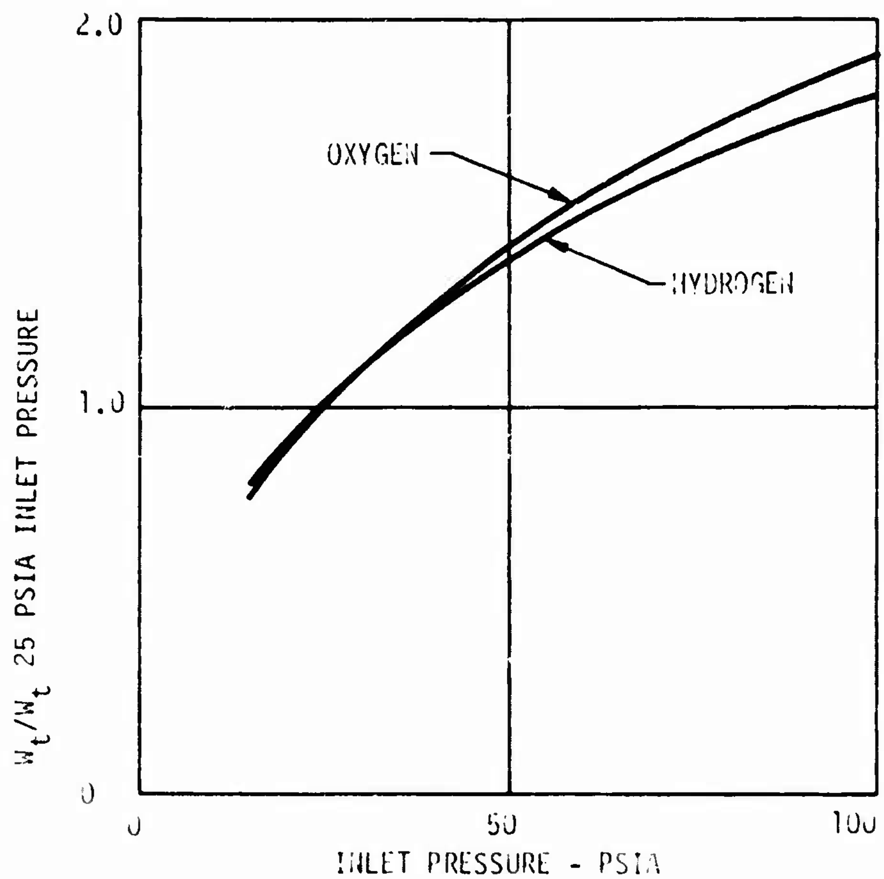


Figure 122. Effect of Inlet Pressure on Pump Chilldown Coolant Weight

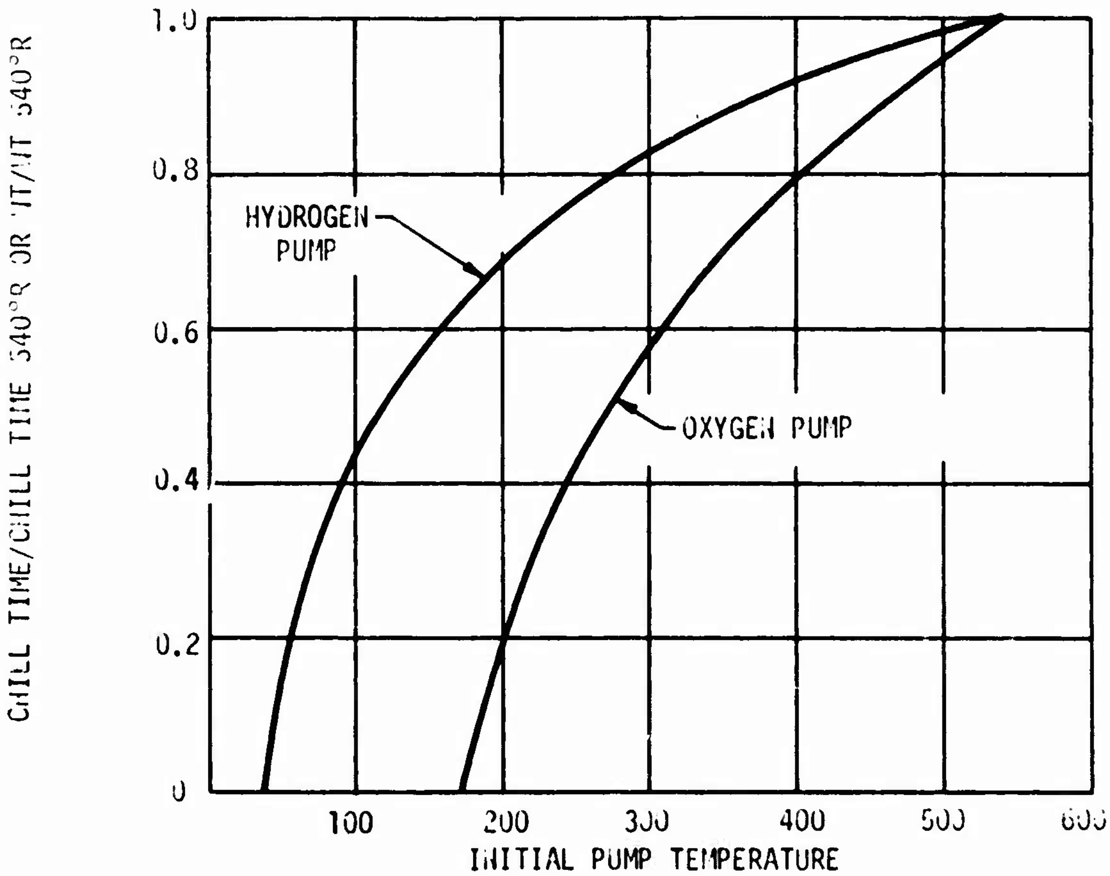


Figure 123. Effect of Initial Temperature on Pump Childown Rate and Coolant Weight

The pump geometry is given in equation form as:

Weight:

$$W_{t_{pump}} = \text{Const} \times \left[\frac{\sqrt{2F/P_c}}{\sqrt{1 + MR} \cdot S \cdot (NPSH + TSH)^{3/4} \sqrt{N}} \right]^{2.5}$$

H_2:Const = 1.99×10^8 O_2:Const = 5.77×10^5

Wetted Area:

$$A_{wetted} = \text{Const} \times \left[\frac{\sqrt{2F/P_c}}{\sqrt{1 + MR} \cdot S \cdot (NPSH + TSH)^{3/4} \sqrt{N}} \right]^{2.0}$$

H_2:Const = 5.18×10^7 O_2:Const = 1.23×10^5

Port Area:

$$A_{port} = \text{Const} \times \frac{F\sqrt{N}}{(1 + MR)\ \sqrt{2\ P_c}}$$

H_2:Const = 0.0136 O_2:Const = 0.01215

In addition it was assumed that film boiling takes place in the passages and that the film coefficient, h_1 could be expressed as:

$$h_1 \sim \text{Reynolds}^{0.8} \sim 1/D^{0.2}$$

A chart solution of the plate problem was utilized, necessitating determination of both the Biot (h L/k) and Fourier (α t/L^2) numbers. Combining the above relationships into an expression for Biot number we have:

$$B_i = \text{Const}\ P_c^{\ 0.3}\ F^{\ 0.15}$$

H_2:Const = 0.066 O_2:Const = 0.226 .

and likewise for Fourier number, solving for time:

$$T_{chill} = \text{Const} \times F_{chill} \sqrt{F/P_c}$$

H_2:Const = 0.00306 O_2:Const = 0.036

Assuming childown is complete when the wall temperature is within 5 degrees of the saturation temperature, the Biot numbers calculated from the above equation are used in conjunction with Chart 25, of P. J. Schneiders' Temperature Response Charts published by John Wiley and Sons, Inc., New York, 1963, to determine the Fourier number and thus chilldown time.

The coolant flow rates, and thus weight for chilldown, were determined assuming choaked vapor flow. The actual flow rates may differ somewhat from those calculated. It is felt, however, that the trends predicted represent reality.

Since the reference values of importance to both chill time and coolant weight appear without powers, both time and coolant weight may be directly proportioned to available data.

(2) Discussion of Results

Chilldown time is presented in Figures 117 and 118. The time increases both with thrust and chamber pressure, reflecting the need for larger pumps port size. It should be noted that the oxygen pump chilldown times are longer than the hydrogen, a result of the much higher hardware mass to wetted surface area ratio given for those pumps.

Figures 119 and 120 show the effect of thrust and chamber pressure on coolant weight used during cooldown. Although coolant weight increases with thrust, it decreases with chamber pressure, reflecting the decrease in port area that occurs as the pump is designed for higher pressure.

Figures 121 and 122 show the effect of pump inlet pressure on chilldown rate and coolant weight respectively. The primary effect is to change the flow rate, thus heat transfer rate.

Figure 123 shows the effect of pump initial temperature on both chilldown time and coolant weight. The time and thus coolant weight decrease sharply with initial temperature.